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FUTURE SPACE TRANSPORTATION SYSTEMS ANALYSIS STUDY

PHASE I TECHNICAL REPORT
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Submitted to
The National Aeronautics and Space Administration
Lyndon B. Johnson Space Center
in Fulfillment of the Requirements
of Contract NAS9-14323

Approved

G. R. Woodcock
Study Manager

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FOREWORD

The Future Space Transportation System Analysis Study, NASA Contract NAS9-14323, is managed by the NASA Lyndon B. Johnson Space Center (JSC) and is being performed by the Research and Engineering Division of the Boeing Aerospace Company in Seattle, Washington. The Contracting Officer's Representative (COR) is Harle L. Vogel of the Future Programs Division of JSC. Study management team members assisting the COR are:

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This document is the technical report for the Phase I study completed May 9, 1975. It has been prepared in such a way that it can be readily updated as additional data are developed in subsequent phases of study.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

NOMENCLATURE

A/B	AEROBRAKING
AL	AUTOMATED LUNAR
ASM	APPLICATION AND SCIENCE MODULE
AU	ASTRONOMICAL UNIT
CAC	CHEMICAL/AEROBRAKING/CHEMICAL
CCC	CHEMCIAL/CHEMICAL/CHEMICAL
CEM	CREW EQUIPMENT MODULE
CONJ	CONJUNCTION
CR	CREW ROTATION
CTM	CREW TRANSFER MODULE
CTV	CREW TRANSFER VEHICLE
EDM	EQUIPMENT DELIVERY MODULE
EEM	EARTH ENTRY MODULE
EM	EQUIPMENT MODULE
EO	EARTH ORBIT
EOI	EARTH ORBIT INSERTION
EOSS	EARTH ORBIT SPACE STATION
FCT	FULL CAPACITY TUG
FM	FLUID MODULE
FVM	FLYING VEHICLE MODULE
GEO	GEOSYNCHRONOUS EARTH ORBIT
GL	GANYMEDE LANDER
GLOW	GROSS LIFT OFF WEIGHT
GSMS	GEOSYNCHRONOUS SATELLITE MAINTENANCE SORTIE
GSS	GEOSYNCHRONOUS SPACE STATION
HLV	HEAVY LIFT VEHICLE
HLLV	HEAVY LIFT LAUNCH VEHICLE
ILSS	INDEPENDENT LUNAR SURFACE SORTIE
IMIEO	INITIAL MASS IN EARTH ORBIT
IUS	INTERIM UPPER STAGE

NOMENCLATURE (Continued)

JP	JUPITER PROBE
JUP	JUPITER
LCHLV	LOW COST HEAVY LIFT VEHICLE
LCOTV	LOW COST ORBIT TRANSFER VEHICLE
LEO	LOW EARTH ORBIT
LFV	LUNAR FLYING VEHICLE
LO	LUNAR ORBIT
LOI	LUNAR ORBIT INSERTION
LRV	LUNAR ROVER VEHICLE
LSB	LUNAR SURFACE BASE
LST	LARGE SPACE TELESCOPE
LSV	LUNAR SORTIE VEHICLE
LTV	LUNAR TRANSPORT VEHICLE
MEM	MARS EXCURSION MODULE
MM	MISSION MODULE
MOI	MARS ORBIT INSERTION
MP	MANNED PLANETARY
MSR	MARS SAMPLE RETURN
MSS	MODULAR SPACE STATION
MSSR	MARS SURFACE SAMPLE RETURN
NNN	NUCLEAR/NUCLEAR/NUCLEAR
NWD	NUCLEAR WASTE DISPOSAL
OLS	ORBITING LUNAR STATION
OMS	ORBIT MANEUVERING SYSTEM
OPP	OPPOSITION
OPS	OPERATIONS
OTV	ORBIT TRANSFER VEHICLE
P/L	PAYLOAD
PM	PROPELLANT OR PROPULSION MODULE

NOMENCLATURE (Continued)

RAM	RESEARCH AND APPLICATION MODULE
RM	RESUPPLY MODULE
RS	RESUPPLY
RVM	ROVER VEHICLE MODULE
SES	SATELLITE ENERGY SYSTEMS
SM	SERVICE MODULE
SOSI	SPACE OPERATIONS AND SCIENTIFIC INVESTIGATIONS
SS	SPACE SHUTTLE
SSE	SOLAR SYSTEM ESCAPE
STG	STAGE
STS	SPACE TRANSPORTATION SYSTEM
TEI	TRANS EARTH INJECTION
TLI	TRANS LUNAR INJECTION
TMI	TRANS MARS INJECTION
USS	UNITARY SPACE STATION

1.0 INTRODUCTION

1.1 STUDY SCOPE

The baseline Space Transportation System (STS), which includes shuttle booster and orbiter and the baseline orbit-to-orbit stage, (either IUS or full-capability tug) will have achieved an initial level of definition by the end of 1974. A number of future space programs, which can be candidates for operations in the post-1985 era, would exceed the expected capability, including any nominal uprating, of the baseline STS performance. *The Future Space Transportation System Analysis Study* is an analysis of candidate future space programs to determine transportation needs, together with a comparative evaluation of alternative ways of evolving future space transportation systems from the baseline STS to meet these needs.

The study includes potential future space programs and missions that could occur in the 1985-1995 timeframe, and are expected to require space transportation capability beyond that of the operational space shuttle, and interim upper stage (IUS). It therefore goes beyond missions included in the space shuttle payloads data bank; such missions generally fall within shuttle/IUS capabilities.

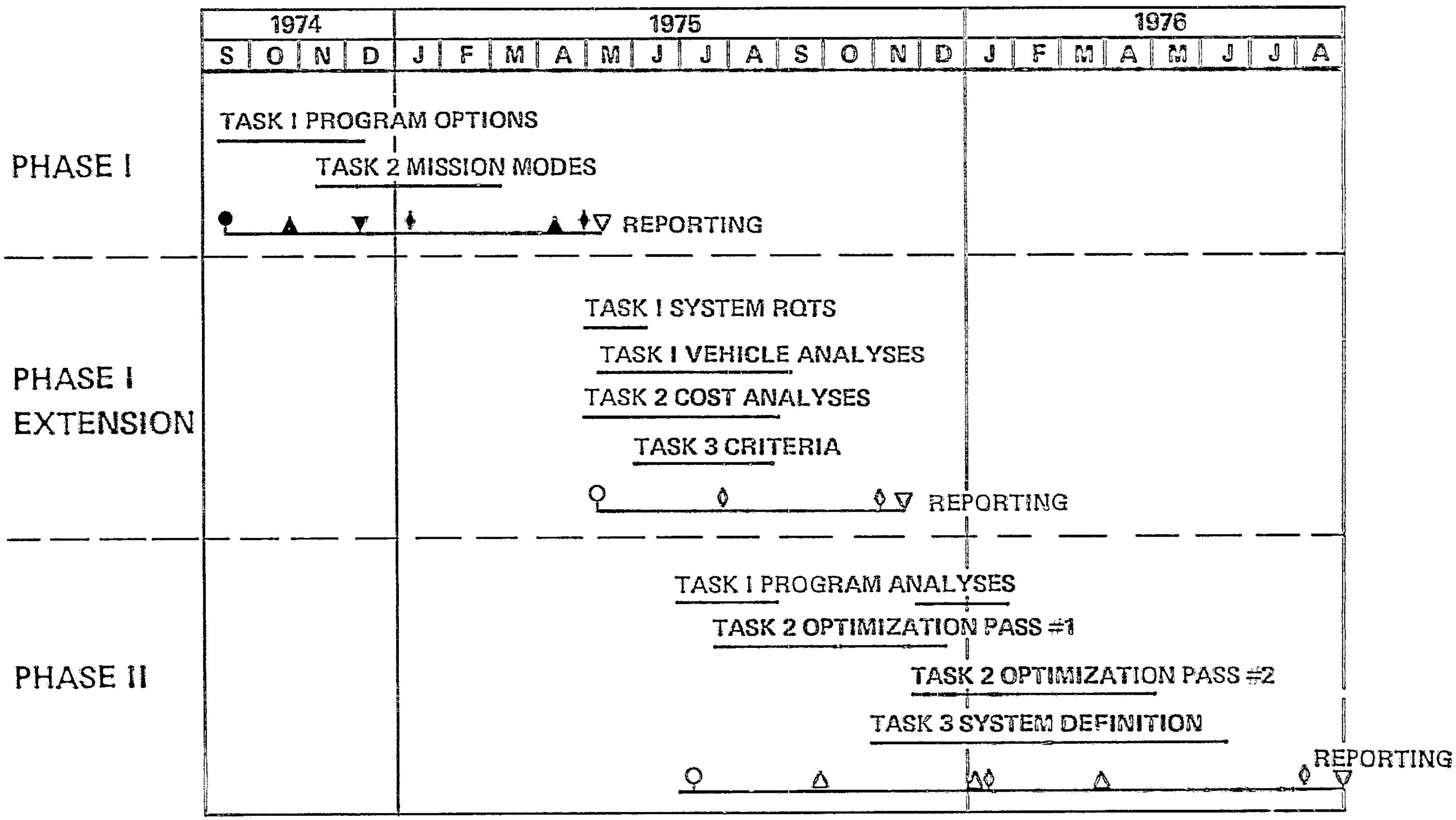
This report describes the results of the Phase I study effort, including space program options and transportation requirements analyses and mission modes and operations analyses. These analyses characterized the requirements of the program options and derived a number of alternative systems approaches to meeting their transportation needs. At this point in the study no attempt has been made to select preferred transportation systems approaches or development strategies. Additional study efforts, presently in the planning stage, will characterize in more detail the technical and programmatic aspects of the transportation systems described herein, and synthesize optimum transportation systems and development strategies for various future program evolution scenarios.

1.2 STUDY OBJECTIVES AND SCHEDULE

The objectives of the study are to define potential transportation requirements and transportation modes for a spectrum of potential transportation activities during the 1985-1995 time period to be used in subsequent definition of an evolutionary space transportation system to satisfy these requirements. Objectives and outputs of the entire study are summarized in Table 1-1. Figure 1-1 shows the schedule, including planning schedules for the phase 1 extension and phase 2 activities.

Table 1-1. Future Space Transportation Systems Analysis Study Overview

	FUNDED		PLANNED	PLANNED
	PHASE 1, TASK 1	PHASE 1, TASK 2	PHASE 1 EXTENSION	PHASE 2
OBJECTIVES	DEFINE THE REQUIREMENTS ON TRANSPORTATION IMPOSED BY POTENTIAL ADVANCED MISSIONS	IDENTIFY REASONABLE ALTERNATE TRANSPORTATION SYSTEM SOLUTIONS TO THE REQUIREMENTS	DEVELOP TECHNICAL AND PROGRAMMATIC CHARACTERISTICS OF THESE ALTERNATIVES DESCRIBE HOW TO SELECT AMONG THEM	SYNTHESIZE OPTIMAL TRANSPORTATION SYSTEMS FOR VARIOUS PROGRAM EVOLUTIONS DEFINE BEST TRANSPORTATION DEVELOPMENT STRATEGIES
OUTPUTS	PROGRAM OPTIONS AND TRANSPORTATION REQUIREMENTS DOCUMENT (INTERIM RELEASED)	MISSION MODE AND OPERATIONS ANALYSIS DOCUMENT: • OPERATIONAL CONCEPTS AND MODES • SYSTEM CONCEPTS • COMPARISONS	TRANSPORTATION SYSTEMS DEFINITION DOCUMENT • VEHICLE CHARACTERISTICS • COST AND PROGRAMMATIC DATA • EVALUATION AND SELECTION CRITERIA	TRANSPORTATION SYSTEM PLANNING DOCUMENT • OPTIMIZED SYSTEMS • DEVELOPMENT PLANS • DEVELOPMENT & EVOLUTION STRATEGIES



O ORIENTATION BRIEFINGS
∇ TECHNICAL REPORTS
◊ FORMAL REVIEWS
Δ WORKING REVIEWS

Figure 1-1. Future Space Transportation Systems Analysis Schedule Overview

1.3 GENERAL GUIDELINES AND ASSUMPTIONS

Five general guidelines were applied during the course of the study:

1) The baseline space transportation system (STS; includes space shuttle and orbit-to-orbit stage, the latter being either IUS or full-capability tug) was assumed operational with capabilities as presently defined by JSC Document No. 07700, Vol. XIV, the Shuttle Payload Accommodations Document. Derivatives of the STS and new systems were analyzed for each program option to the extent that additional capabilities were required.

2) Program options were analyzed separately and independently. This means that systems from one program option, e.g., low Earth orbit space station, were not assumed available to support another option, e.g., lunar base. Further, no attempt was made to force common transportation solutions across programs although after-the-fact commonality comparisons were made.

   A later phase of study (Phase II), will address these questions. They are to be addressed as a departure from tailored solutions in order that the true impact and cost of commonality can be assessed against its benefits.

3) Technologies assumed for program options and for transportation systems developments were limited to those that could be forecast by firm technical extrapolation of present knowledge. This is considered appropriate for the time frame of principal interest, 1985-1995 and for the purposes of the study. Systems operational in 1985 are likely to incorporate only advances in the state of the art founded on base technology firmly in hand today. These, of course, can result in significant performance improvements over currently operational systems. By 1995, systems may become operational with major, even unforeseen technical advances. No attempts were made to forecast these and no dependence on such breakthroughs was utilized in predicting system performance.

4) Program options selected for study were considered to be technically feasible in the 1985-1995 time frame. These selections were not intended as predictions.

5) New systems capabilities were analyzed only to the extent that a need or potential need could be perceived from the Task 1 (program options and transportation requirements analysis) results. The lack of discussion of particular types of advanced systems does not signify a judgement by this study activity regarding their feasibility, but rather that a need for that capability was not perceived within the scope and level of detail of the potential future programs analyzed.

2.0 STUDY APPROACH AND STATUS

Development of the results presented herein took place in four distinct steps:

1) Descriptions of each program option were developed from results of prior NASA studies, other sources, and from new work where required. These descriptions represent a requirement for transportation, i.e., a set of problems needing solutions.

2) Requirements analyses developed a set of payload delivery options and requirements, that is, requirements on transportation as a basis for development of solutions.

3) Transportation/mission modes and operations analyses developed a number of transportation system options and sequences for each program/mission option.

4) Qualitative comparisons and evaluations developed pro's and con's and assessments of practicality for each option.

Results of each of these steps are described for each program/mission option in the main body of this document. Supporting analyses and data are presented in the appendix (volume 2).

2.1 POTENTIAL FUTURE SPACE PROGRAM OPTIONS

Nine program options were specified by the NASA Statement of Work, and another was added early in the study. They are summarized in Table 2-1. All of these options have been studied by prior NASA studies -some quite extensively. Task 1 of this study utilized the results from these prior studies as applicable. Reference documentation is identified throughout the body of this report. A complete list of references appears at the end of the report. New analyses and concept studies were conducted as necessary to characterize transportation requirements not available from the references. Missions characterized by new analyses are underlined in the table.

One of the elements of task 1 was a search for new program options to be considered for analysis by the study. Seven options were investigated; five were retained. The two program options not included (space colonization; interstellar probes) were considered to represent technical developments most likely to occur later than the timeframe (1985-1995) under study. Four of the five new options were found to be best treated as optional features of the program options specified by the Statement of Work; three of these required concept analyses to achieve sufficient definition for purposes of this study.

Table 2-1. Program Options

Program	Missions	Objectives
1. Low Earth Orbit Space Stations	• 12-man modular or unitary station • 60-man space base	• Broad spectrum earth observatory • Develop space manufacturing • Scientific investigations
2. Geosynchronous Operations	• 8-man modular or unitary station • Satellite maintenance sortie	• Earth observations • Communication/navigation • Maintenance and repair of automated spacecraft
3. Independent Lunar Surface Sorties	• 4-man self supporting landing	• In-depth exploration of selected areas
4. Orbiting Lunar Station	• 8-man modular or unitary station with surface sortie	• Broad spectrum surface observation • 4-man, 28-day sorties
5. Lunar Surface Base	• 6-man, 6-month • 12-man, semipermanent	• Astronomical observations • Surface exploration • Indigenous material utilization

Table 2-1. (continued) Program Options

Program	Missions	Objectives
6. Manned Planetary	• Manned Mars landing • Opposition • Conjunction • Venus swing-by	• 3-man, 30 day sortie • Planetology • Effects of modifying forces • Search for life forms
7. Automated Lunar	• Orbital observatory • Backside lander • Relay satellite	• Broadband scientific observation • Long duration Rover with sample return
8. Automated Planetary	• Mars lander • Jupiter atm probe • Ganymede lander	• Rover/sample return • Invest upper cloud system • Orbital observation and surface sample analysis
9. Nuclear Waste Disposal	• Refined waste • Total waste	• Permanent waste disposal
10. Satellite Energy Systems	• On-orbit power generation • On-orbit power reflectors	• Commercial electric power • Long range power transmission

The remaining new option, representing an addition to the list, was manned planetary exploration. Manned planetary missions have been extensively studied by NASA and a considerable data base exists. It was therefore deemed appropriate to evaluate the applicability of transportation system options developed by this study for the other program options to manned planetary missions.

Program option descriptions in this document are representative of the program options in terms of transportation requirements. They are not recommendations or conclusions regarding objectives or system design approaches. A variety of implementations could be applied to these program options. This is particularly true of the automated planetary option. It was impractical to treat comprehensively the large number of potential destinations and mission types for automated planetary exploration. The three missions selected are indicative of the range of transportation capability needed.

2.2 TRANSPORTATION SYSTEMS INVESTIGATED

Transportation systems of three general operational capabilities were analyzed: Earth launch systems, orbit transfer (orbit-to-orbit) systems, and lunar/planetary transport systems. Operational relationships are schematically depicted in Figure 2-1. This functional division maximizes efficiency of the classes and types of transportation systems considered in this study.

2.2.1 Earth Launch Systems

2.2.1.1 Space Shuttle

The space shuttle presently under development by the NASA (Figure 2-2) was considered the baseline Earth launch vehicle in this study. Its characteristics and capabilities were obtained from JSC 07700, Vol. XIV, the payload accommodations document.

2.2.1.2 Heavy Lift Vehicles

Heavy lift vehicle systems were characterized for delivery, to the Earth orbit interface, of payloads too massive or too large in volume for the shuttle payload bay. Heavy lift vehicles may be expendable, partially reusable, or reusable, and single or multiple stage. Two classes of heavy lift vehicles were considered; those derived from Space Shuttle elements and those largely or entirely new systems. The shuttle derivatives are applicable to all programs except those (e.g., satellite power stations) requiring low recurring costs achievable only through total reusability. Two shuttle derivative options were considered: an all-solid rocket booster (SRB) configuration and a configuration using either two or four SRB's clustered around a modified shuttle external tank (ET) made into a stage, fitted with three space shuttle main engines (SSME's) in a recoverable propulsion

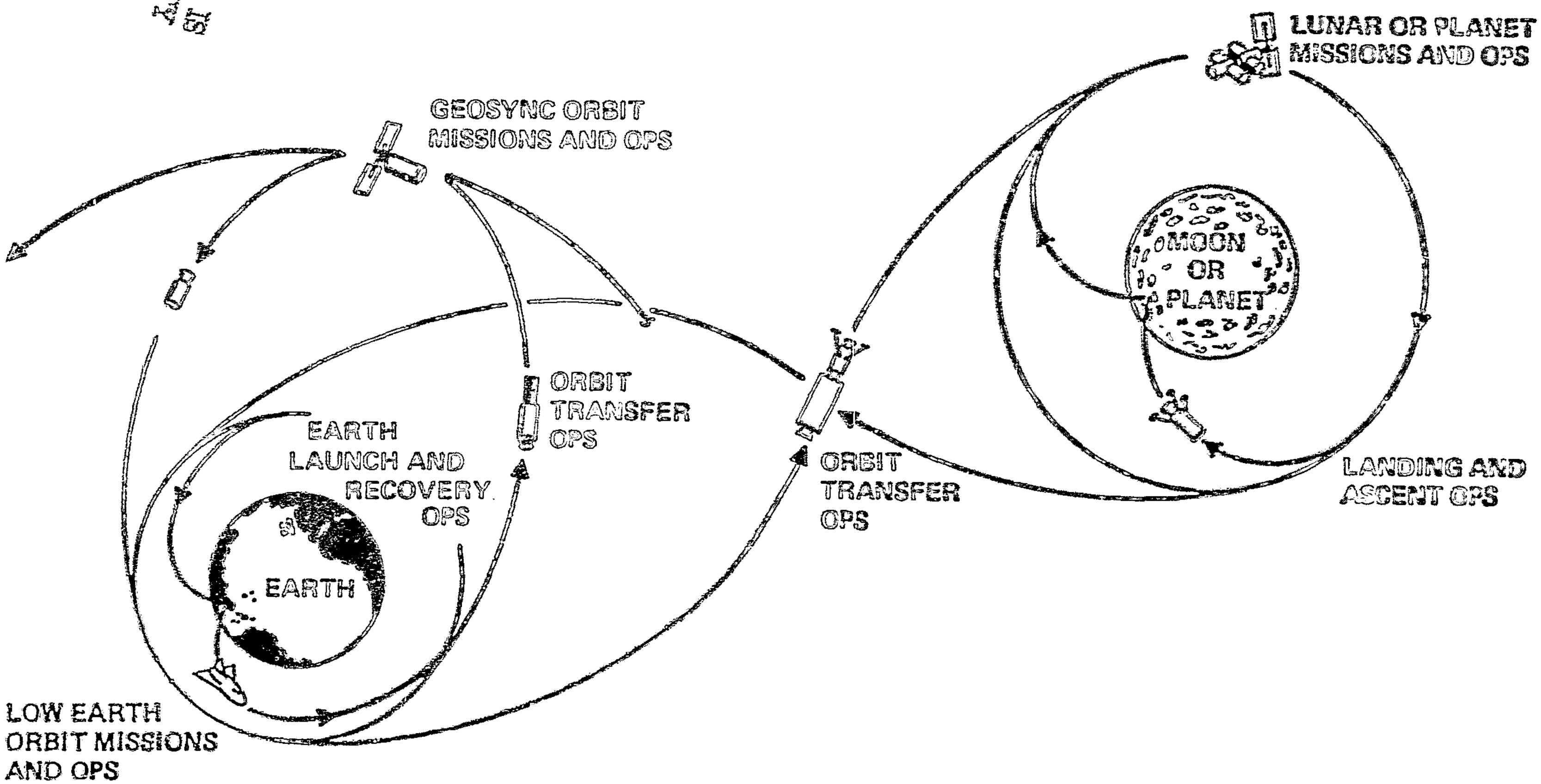


Figure 2-1. Mission Transportation Operations

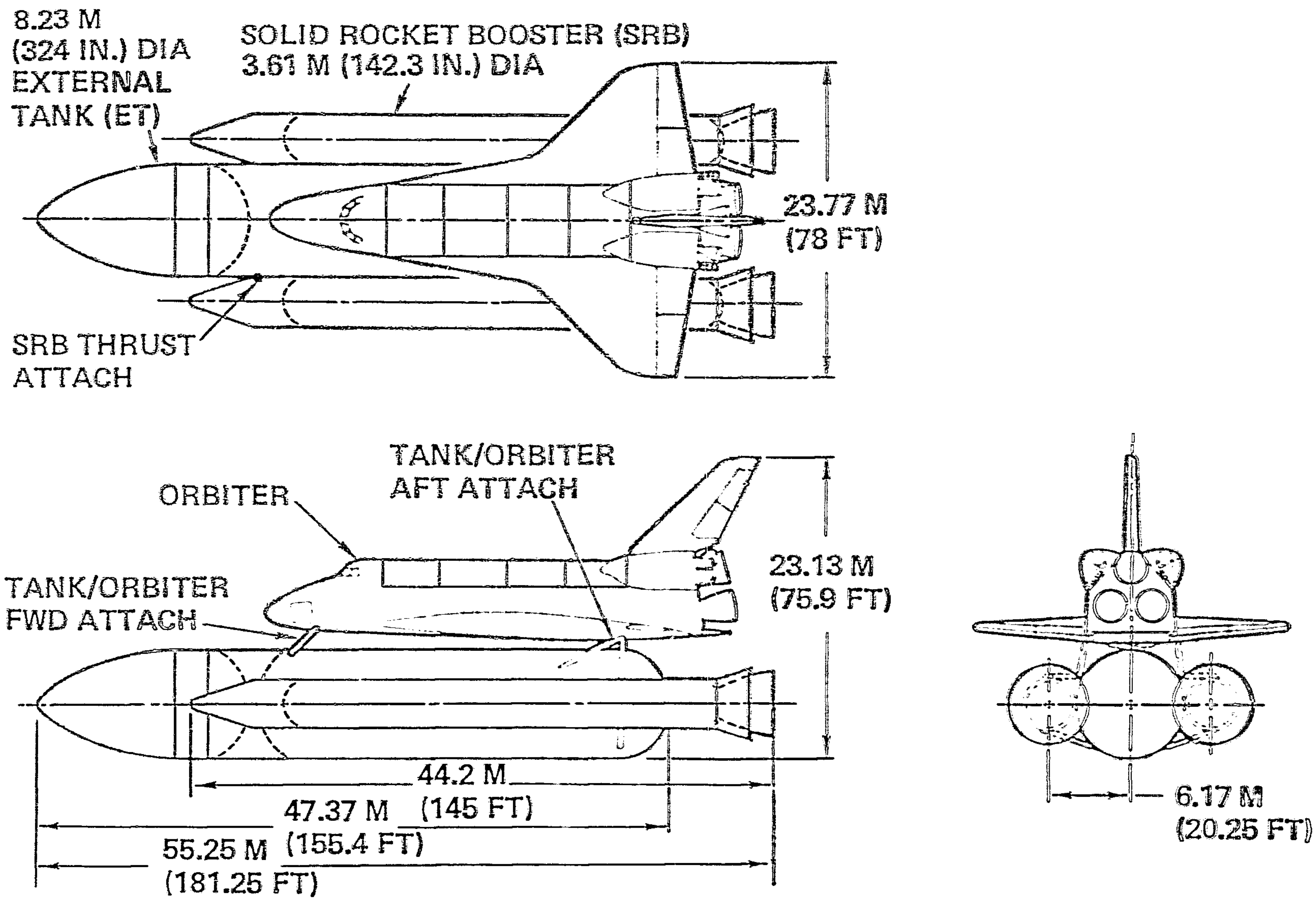


Figure 2-2. Space Shuttle Vehicle

package. The SRB/ET system was generally more suited to the missions studied, due to its greater payload capability. A representative low cost heavy lift vehicle was adopted from a Boeing IR&D study of power satellite launch systems. It is one of many potential ways to achieve very low cost through total reusability.

The Earth launch options are compared in Figure 2-3.

Earth launch systems were not tailored to each program/mission option; the applicable ones from the foregoing options were utilized as appropriate.

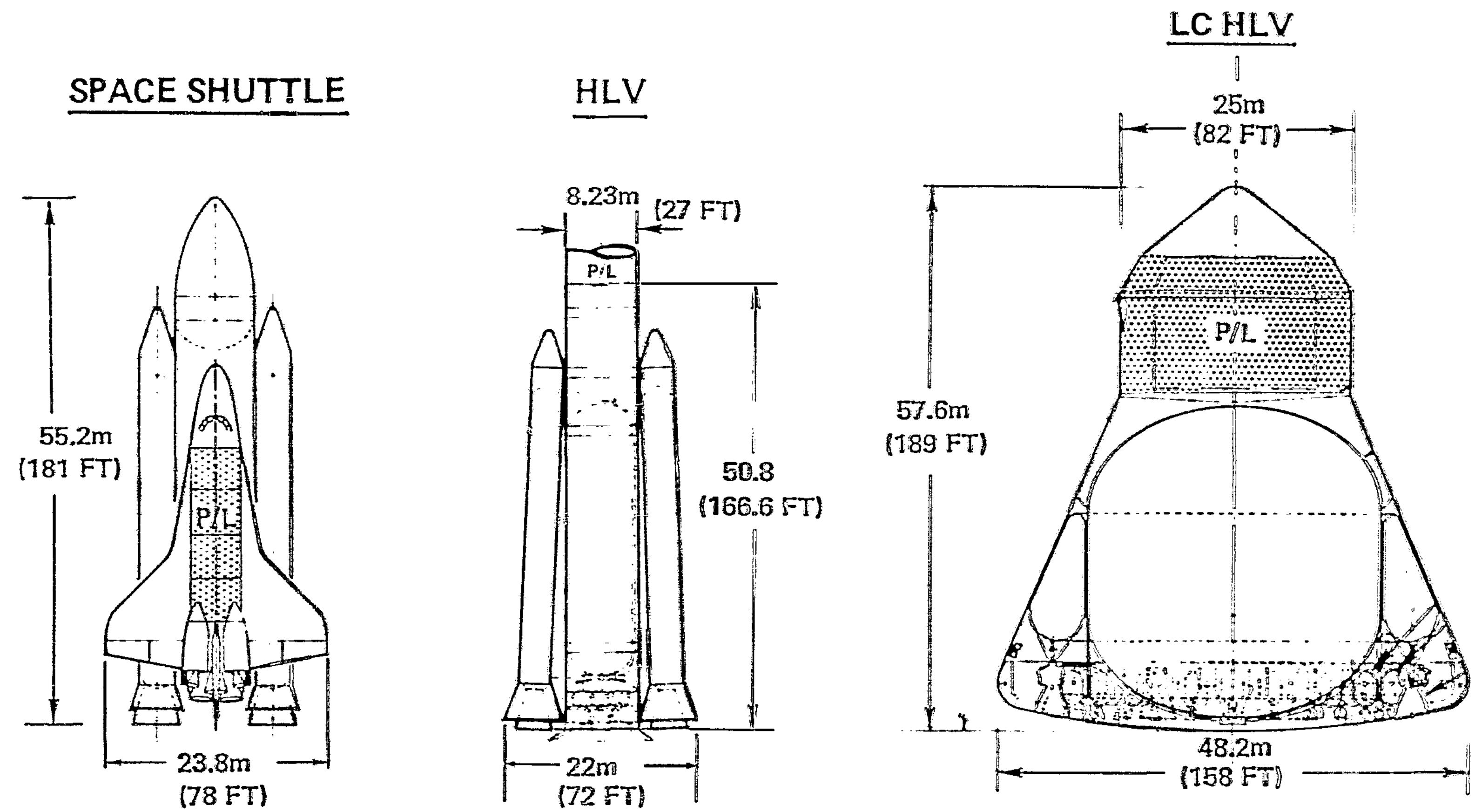
2.2.2 Orbit Transfer Systems

Orbit transfer systems provide the necessary propulsive impulse to transport payloads from a low Earth orbit operational interface to transfer and mission orbits. In some cases the mission orbit itself is an interface between orbit transfer operations and lunar or planetary surface transportation. Orbit transfer systems include be high thrust and low thrust systems.

2.2.2.1 High Thrust Systems

High thrust systems utilize (as propellant) the working fluid of the thermodynamic cycle that converts source energy (chemical or nuclear) into thrust power. This study considered liquid oxygen with liquid hydrogen (LO_2/LH_2) and liquid oxygen with monomethyl hydrazine (LO_2/MMH) as chemical propellants, and nuclear reactor heated LH_2 as propulsion systems. Many alternative chemical propellants could be employed. LO_2/LH_2 provides high performance at low bulk density. A representative specific impulse is 4 500 m/sec (459 seconds) in an orbit transfer application. (The specific impulse in SI units is the effective jet velocity). The high performance of LO_2/LH_2 minimizes the mass that must be transported to the low orbit operational interface but the low density of LH_2 results, in many cases, in transportation stages too large for the shuttle payload bay (4.57 x 18.29m; 15 x 60 ft). LO_2/MMH provides less specific impulse performance; a representative value is 3 630 m/sec (370 sec). Its comparatively high density results in stages that can be fit within the shuttle payload bay for launch or recovery. The mass of propellant required is much greater than shuttle capability; on-orbit fueling is required. Nuclear reactor heated LH_2 provides steady-state specific impulse on the order of 8 090 m/sec (825 sec) but at the very low propellant density of LH_2. Propellant losses for start-up and afterheat removal result in an effective Isp for the LH_2 nuclear rocket of about 7 650 m/sec(780 sec).

High-thrust systems can be operated in a variety of staging modes to improve performance, transportability, or operational flexibility. The staging modes considered in this study are illustrated in Figure 2-4. They have the following principal features:



	Space Shuttle		HLV		LC HLV
• PAYLOAD @ 270 NM 28.5 DEG	25 000 KG* (55,000 LB)*	W/2 SRB	72 270 KG (159,000 LB)		250 000 KG (550,000 LB)
		W/4 SRB	110 910 KG (244,000 LB)		
• GLOW (10^6)	1.91 KG (4.2 LB)	W/4 SRB	3.2 KG (7.09 LB)		10.27 KG (22.6 LB)

* WITH ONE OMS KIT

Figure 2-3. Earth Launch Vehicle Candidates

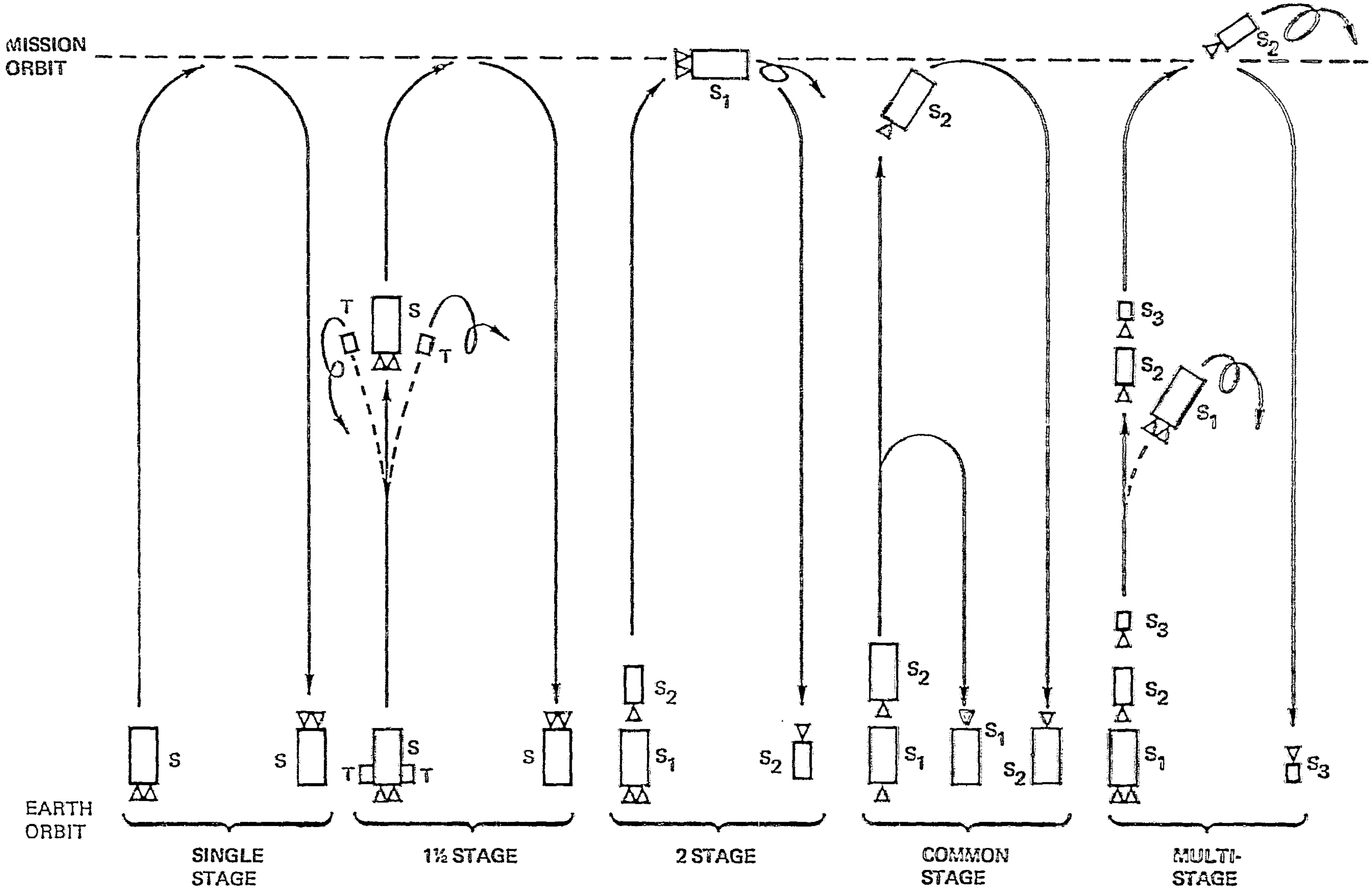


Figure 2-4. OTV Staging Modes

- The single stage mode is operationally the simplest. The single orbit transfer stage executes the entire mission from low Earth orbit to destination (and return if applicable). It is normally reusable; the stage returns itself to the low Earth orbit interface even if the mission function requires payload delivery only. The single-stage vehicles that resulted from analyses in this study were too large for return to Earth by the shuttle; accordingly they must be refueled and serviced at the low Earth orbit interface.

- The 1-1/2 stage mode augments the propellant capacity of the propulsive stage with auxiliary containers (drop tanks). The drop tanks are jettisoned when their propellant (used first) is expended; they are not recovered. This mode reduces initial mass by virtue of the efficiency improvement resulting from staging. Further, the division of required propellant between two or more containers allows each individual container to be size-compatible with the shuttle.

- The conventional two-stage mode provide an expendable stage for delivery and a reusable stage for return where return is needed. The two stages are much different in size, the expendable stage being the largest.

- The common stage mode, sometimes called tandem or slingshot, defines the two stages as having equal propellant capacity and divides the transportation task accordingly. The first stage, expending about 90% of its propellant, is able to boost the second stage and payload to an elliptic Earth orbit. It then returns to the low Earth orbit for reuse, using its remaining propellant. This boost reduces the delta V required of the upper stage by typically 2,000 m/sec (6,562 ft/sec). The actual delta V split depends on the mission and payload. This staging mode is about as efficient as the others mentioned because the upper stage is large and therefore has a higher propellant fraction than is otherwise possible. Both stages are normally reused. LO_2/MMH orbit transfer vehicles (OTV's) were examined only in this mode.

- Multistage modes were examined for missions requiring very high total delta V, such as planetary missions. One of the options examined for the Independent Lunar Surface Sortie mission was a multistage mode.

The baseline STS includes two OTV options; the interim upper stage (IUS) for which several potential configurations are under study, and the full capability tug (FCT) defined by MSFC as illustrated in Figure 2-5. The FCT is quite similar to some of the smaller OTV options considered in this study.

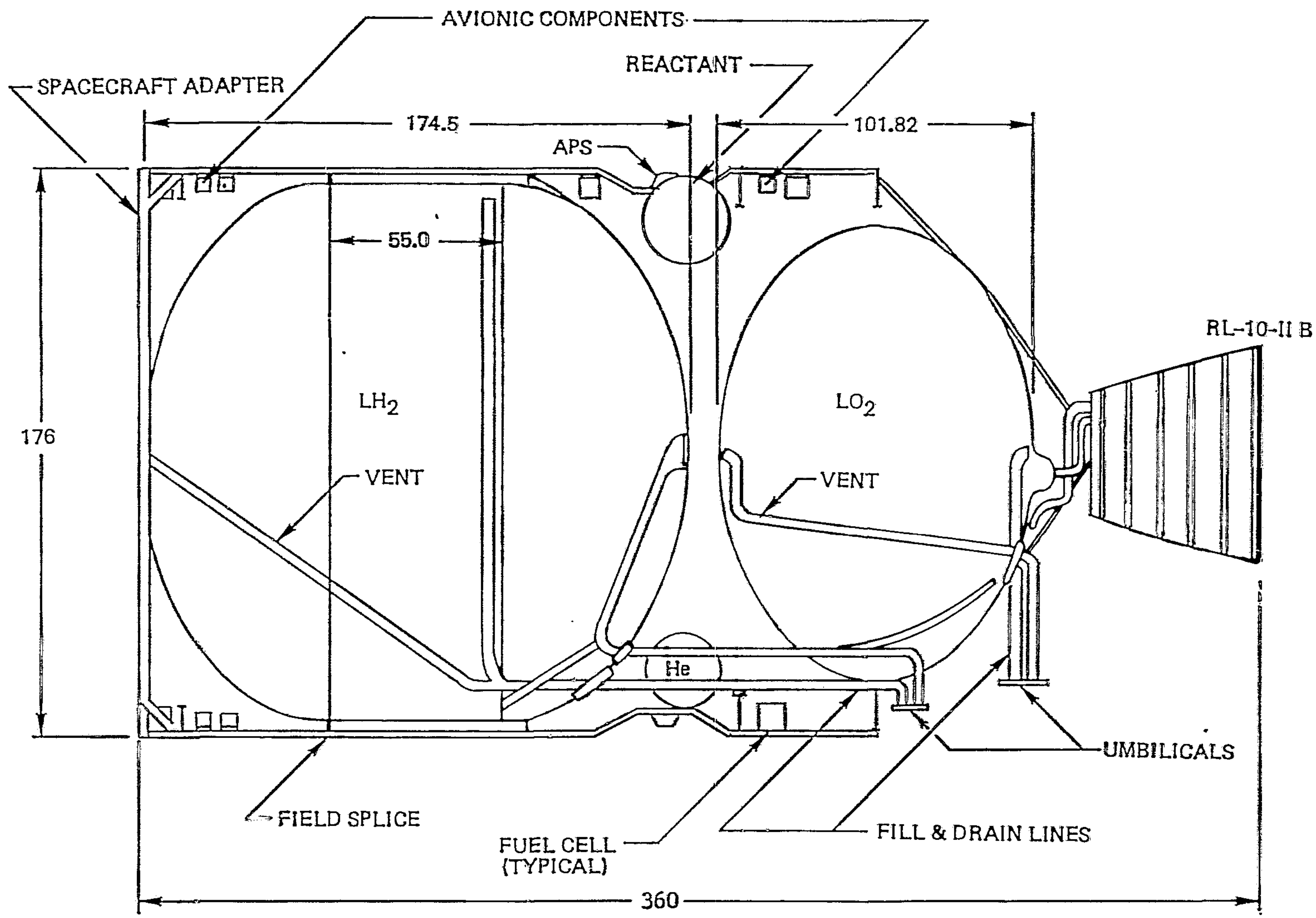


Figure 2-5. Full Capability Tug (MSFC)

2.2.2.2 Low Thrust Systems

Low thrust systems examined in this study consisted of power generation systems (solar photovoltaic, solar thermal, or nuclear reactor thermal) driving thrust producing systems (electric thrusters) that produce a jet from a propellant not involved in the power production or transmission processes. These systems are low thrust because the specific masses of power generation systems, i.e. in kg or lb per kilowatt, are large in comparison with the thrust that can be produced by their output, i.e., in newtons or lb per kilowatt. Acceleration levels are typically limited to less than 10^{-2} m/sec^2 (10^{-3} g's) and thrust is typically applied continuously throughout the orbit transfer. Because the thrust is generally produced by electrical or magnetic body forces on the propellant rather than a temperature-limited thermodynamic expansion, very high specific impuse can be attained. 25 000 m/sec (2,550 sec) is a representative value, but specific impulse can be selected over a relatively wide range to suit particular mission tradeoffs between trip time, inert mass, and propellant mass requirements. This tradeoff is particularly important in the power satellite missions.

Low thrust systems were generally examined in single-stage modes. Staging of a sort was considered for the power satellite mission. Thermal-engine power satellites are capable of providing the power for the electric propulsion system on ascent. In this case, the low thrust vehicle may consist only of power conditioning, control, thrusters, and propellant tankage. A potentially desirable mission mode involves return of the high-cost elements to low orbit with a highthrust stage (LO_2/LH_2) and disposal of the ascent propellant tanks, leaving them in the high orbit. These alternatives are discussed in more detail in Section 3.10.

The low thrust systems required for missions in this study are much higher in power but otherwise similar to the SEPS concept depicted in Figure 2-6.

2.2.3 Lunar Transport Vehicles (LTV's)

Lunar transport vehicles operate between the lunar orbit interface and the lunar surface. They are similar to high-thrust chemical OTV's and are equipped with landing equipment including landing legs, landing and targeting avionics including the required guidance and navigation software and radar, throttleable engines, and crew aids including ladders and/or elevators, etc. Commonality between LTV's and OTV's may be more apparent than real since landing/ascent operations impose a number of unique design requirements. This issue will be addressed in a later phase of study. Single stage, 1-1/2 stage, and two stage options were examined for LO_2/LH_2 and a single stage LO_2/MMH vehicle was considered. Low thrust vehicles are not capable of lunar or planetary landing.

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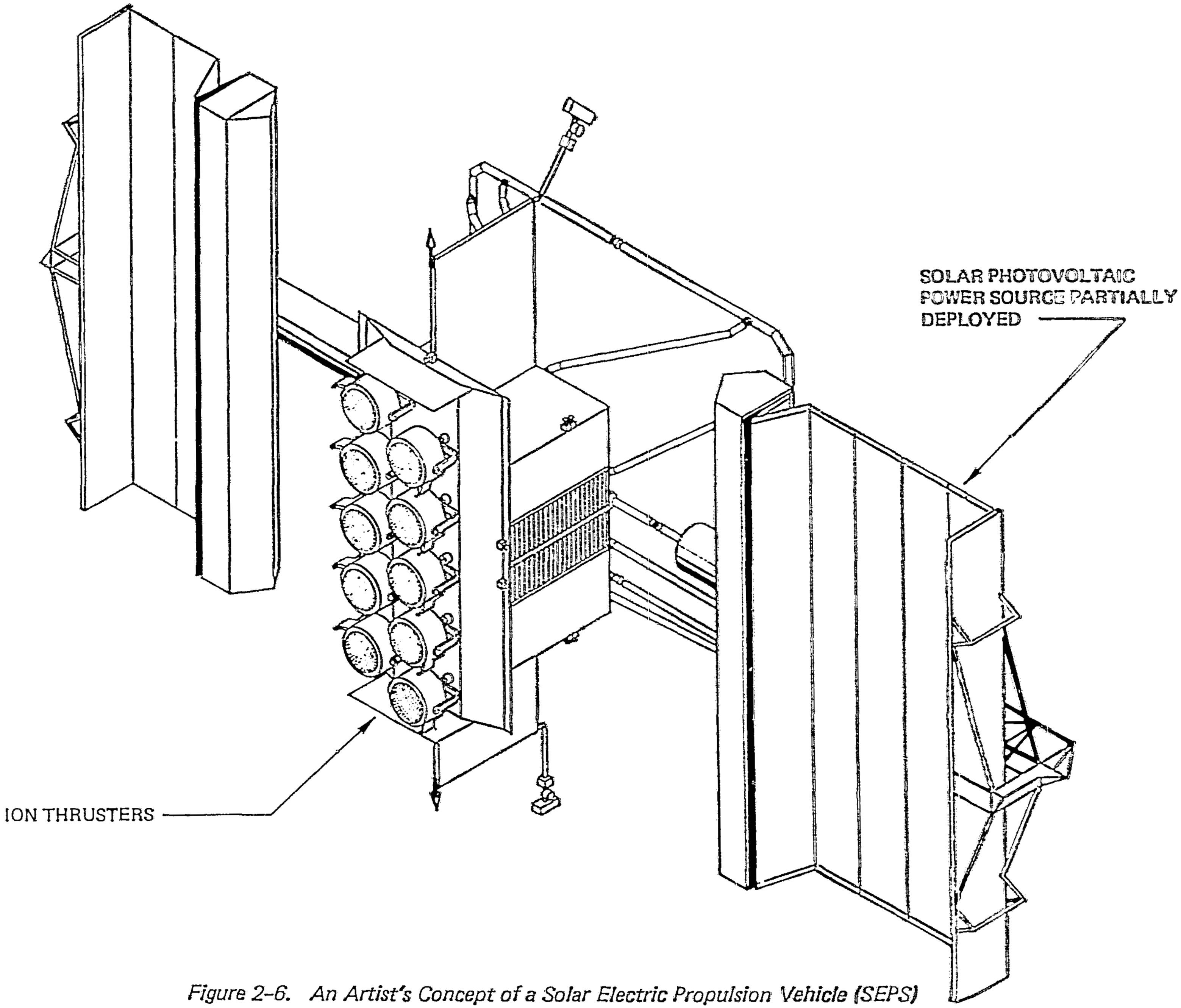


Figure 2-6. An Artist's Concept of a Solar Electric Propulsion Vehicle (SEPS)

An analogous requirement for planetary landing exists but was given relatively little emphasis in this study. Automated Mars landers and a Ganymede lander were characterized as mission payloads. Manned Mars landers were drawn from prior studies and also treated as mission payloads.

2.3 SUMMARY OF KEY RESULTS TO DATE

This summary includes an overview of requirem ts represented by the program options and of transportation system options. Emphasis is placed on comparison. Each program/mission option is discussed individually in detail in the body of the report.

2.3.1 Requirements Drivers and Ranges

Requirements drivers include—

- Mass of individual payload items
- Size of individual payload items
- Energy required for payload delivery
- Total mass to be delivered
- Mass delivered per year required to sustain the programs
- Economic constraints on systems (such as power production) intended to provide or contribute to a commercial product.

Masses and sizes of the largest individual payload items for each program option are shown in Figure 2-7. Most of the payloads from past studies were defined as compatible with space shuttle launch, at least in a modular option. Only the space base and some of the planetary mission payloads cannot be transported to low Earth orbit by the shuttle. The space base is a large low-orbit space station for a crew of 60 or more. It is assembled from modules derived from the unitary space station. These modules are roughly 9 meters (30 ft) in diameter by 20 meters (65 ft) in length; various sizes in this range have been studied. Assembly of a space base from shuttle-sized modules would require approximately 50 modules; practicality of this has not been investigated. Propellant is included in the largest individual masses for the manned lunar missions. For shuttle-compatible launch, the propellant can be offloaded.

Manned netary missions and some of the automated planetary options include a Mars landing vehicle too large for the shuttle. The manned planetary mission also includes a large mission module. The mission module could presumably be assembled from modular space station elements. Modular construction is impractical for the Mars lander; it requires a large diameter to effectively accomplish aerodynamic braking during entry into the tenuous Mars atmosphere.

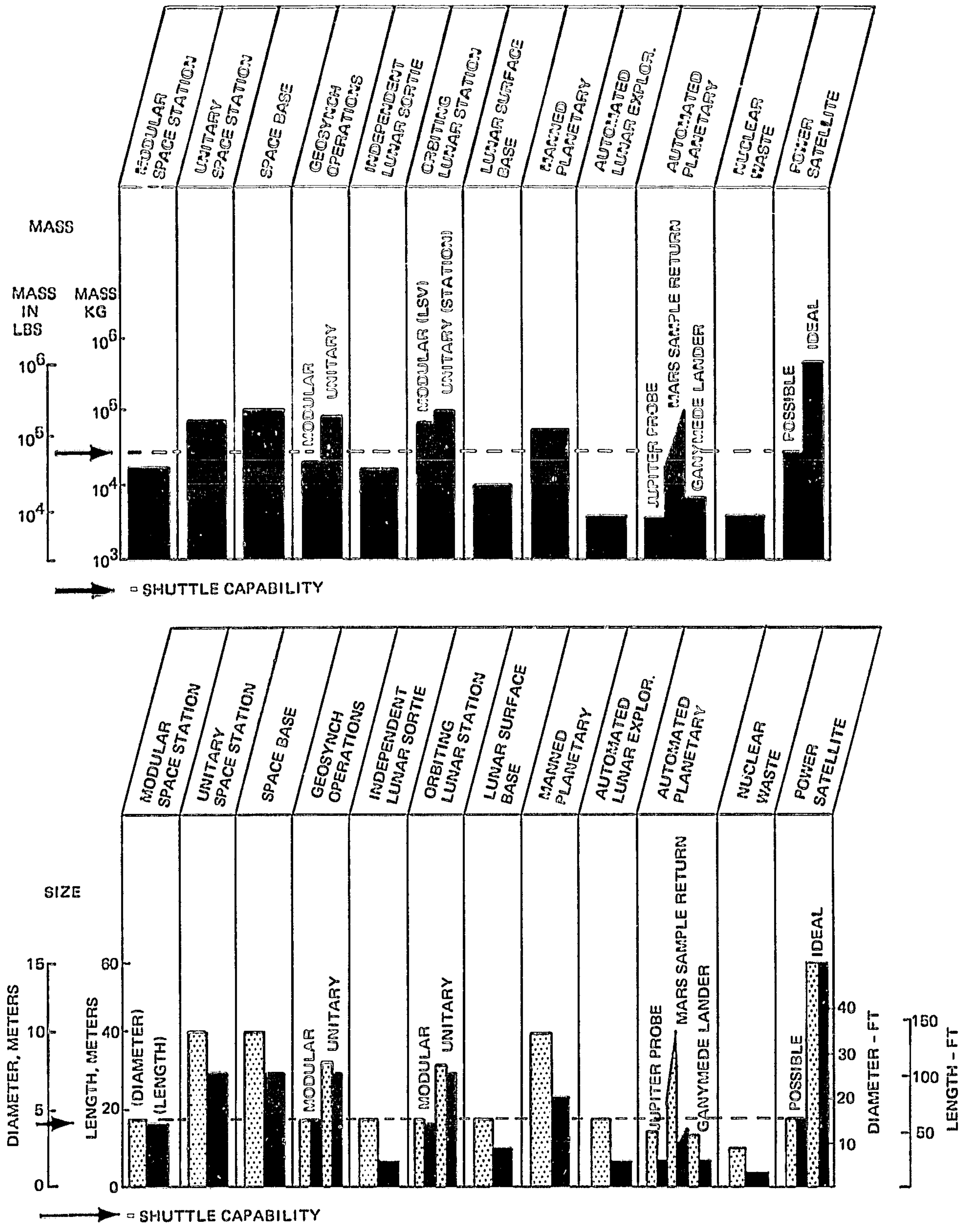


Figure 2-7. Largest Individual Payload Items

Power satellites could conceivably be assembled in a low orbit from components compatible with shuttle launch. However, orbital assembly of the powersat is greatly facilitated by launch in larger, lower density packages.

Figure 2-8 represents a comparison of total payload mass delivery requirements. Annual payloads are stated only when a valid technical basis exists for continuing delivery requirements. For example, in the case of the automated lunar program, as many or as few missions as desired may be flown in any given time period; an annual requirement is not stated.

The relatively large delivery requirement for the orbiting lunar station (OLS) results because one surface sortie occurs after each resupply flight (i.e., 3.35 sorties per year). The OLS program as defined involves more trips to the surface than the lunar surface base program (2.23 resupply and crew rotation flights to the lunar surface per year). It should be recognized that the frequency of surface sorties for the OLS mission is not a definite number; more or fewer than 3.35 per year could be conducted.

Two of the program options analyzed must be economically compatible with the commercial production of electric power. It is envisaged that the power satellite will produce commercial power. Nuclear waste disposal (as currently proposed) will be accomplished by launching the radio-active wastes from commercial power reactors to a final disposal in space. Costs for this would appear as a surcharge on power production costs. Economic requirements can therefore be approximated for the transportation systems for these program options. The resulting economic requirements are directly related to the nature of proposed transportation solutions and are discussed in the body of the report under the pertinent program options.

2.3.2 Generic Transportation Needs and Technology

2.3.2.1 Generic Needs

Starting with the current STS as a baseline, four generic types of additional needed transportation capability were identified by Task 1:

- Heavy Lift—The capability to launch, from Earth's surface to a low Earth orbit, payloads and transportation stages too heavy or too large in volume to be accommodated in the payload bay of the shuttle.

- Orbit-to-Orbit Transfer—The capability to deliver payloads from a low Earth orbit to a mission or destination orbit. In many cases a return capability is also required.

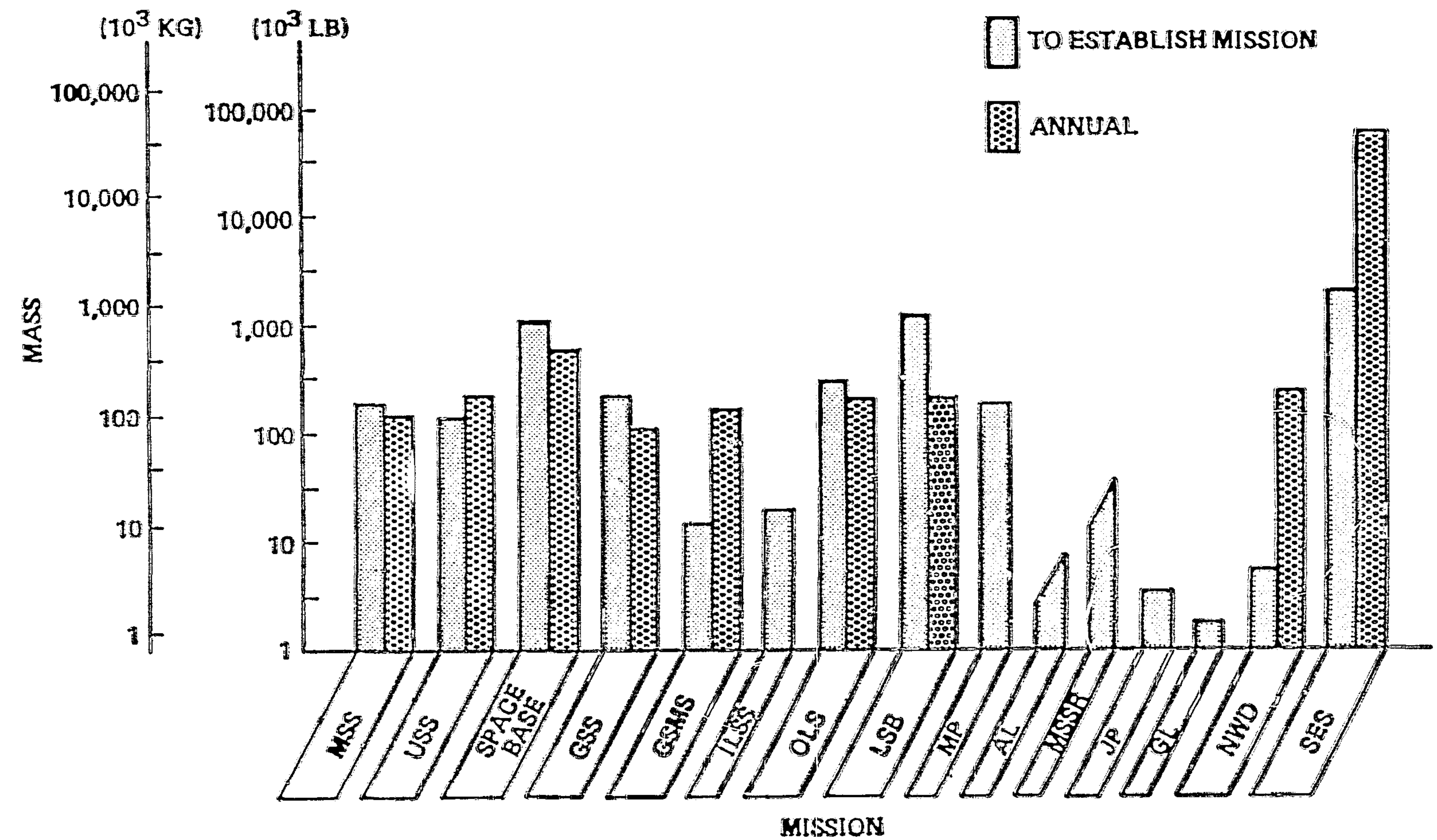


Figure 2-8. Mission Imposed Transportation Requirements Payload Only

- **Lunar or Planetary Landing** The capability to land space systems and mission crews on the moon or on Mars and to return the crews to an orbit for subsequent return to Earth.
- **Low Cost Operation** The capability to conduct space transportation operations at very low costs in order to support missions requiring movement of great masses.

The existence of these types of needs was confirmed by Task 2; no additional types of needs were found.

Based on these generic types, the missions analyzed may be placed into six categories as to their basic transportation needs. These categories are summarized in Table 2-2. It should be noted that although lunar/planetary landing is a "category," the characteristics of vehicles in this category are highly dependent on the characteristics of the particular lunar or planetary body for which they are designed. The following observations are made:

- All but three of the missions can be done using the shuttle as the Earth launch vehicle, but advanced orbit-to-orbit transportation is required in most cases due to the magnitude of masses to be transported.
- The heavy lift vehicle is a key decision. Its effects on other system elements are highly significant: a) it frees the mission payloads, particularly manned station elements, from the size constraints of the shuttle payload bay (but the large modules cannot be returned to Earth); b) it allows the use of larger, potentially more efficient orbit-to-orbit stages (but they also cannot be returned to Earth); c) probably most significant is its reduction in the numbers of Earth launches required to accomplish the more demanding missions.
- The three missions (manned planetary, nuclear total waste disposal, and power satellites) that cannot be accommodated by the shuttle will require tailoring of transportation. The degree of tailoring for manned planetary will, as described below, depend on the approach to the mission. Present indications are that the other two missions need, and justify, systems economically optimized for their unique requirements.

2.3.2.2 Technology Needs

Four areas of technology needs were observed frequently and warrent consideration for technology or study activities:

- Docking together in orbit of massive vehicles or of vehicles to payloads in order to form complete transportation/payload units. Masses are in the range of 50 000 kg to 300 000 kg (110,000 lb to 660,000 lb) depending on the modes and sequences selected. In most cases a large proportion of the mass is liquid propellant.
- Transfer of propellant on orbit, typically from a tanker to a transportation stage.

Table 2-2. Transportation System Summary

	SHUTTLE OR SHUTTLE + IUS OR FCT	SIGNIF BENEFIT FROM HLV	HLV	OTV	LTV/PTV	LGHLV AND LGOTV
MODULAR LOW ORBIT SPACE STATION	X					
UNITARY LOW ORBIT SPACE STATION & SPACE BASE	X		X			
GEOSYNCHRONOUS SATELLITE MAINTENANCE	X			X		
GEOSYNCHRONOUS MODULAR STATION	X	X		X		
GEOSYNCHRONOUS UNITARY STATION	X		X	X		
INDEPENDENT LUNAR SURFACE SORTIES	X	X		X	X	
MODULAR ORBITING LUNAB STATION	X	X		X	X	
UNITARY ORBITING LUNAR STATION	X		X	X	X	
LUNAR SURFACE BASE	X	X		X	X	
MANNED PLANETARY LANDING	X		X	X	X	
AUTOMATED LUNAR	X				X	
MARS SAMPLE RETURN	X	X		X	X	
JUPITER BUOYANT PROBE	X	X		X		
GANYEME LANDER	X			X	X	
NUCLEAR WASTE DISPOSAL-REFINED WASTE(TOTALWASTE)	X	X				(X)
POWER SATELLITES						X

- Availability of a space engine in the thrust range 80,000 to 200,000 N (18,000 to 45,000 lb), with high Isp, tank head ullaging and start, long life with no limit on restarts, and compact packaging. Where used for lunar or planetary landing, throttling is required. LO_2/LH_2 or LO_2/MMH are candidate propellants; the choices between these and others will be addressed in a later phase of the present study.
- Availability of an electric thruster with the following general characteristics:
  a) Efficiency better than 50% from busbar power to effective jet power.

  b) Common material such as water, ammonia, sodium, or hydrogen for working fluid.

  c) Lightweight and compact packaging–less than 1 kg/kw (2.2 lb/kw) of electric power consumed including whatever power conditioning is required.

  d) Isp in the range of 25 000 m/sec (2,500 sec) and higher.

The fourth technology need arises for the missions requiring very low cost transportation. The others are present in most of the potential programs analyzed.

2.3.3 Earth Launch Systems

2.3.3.1 Space Shuttle

As stated above, the space shuttle provides a sound basis for advanced space transportation and fills a large proportion of the potential Earth launch requirements. A number of payload requirements were observed in the mass range 25 000 to 45 000 kg (55,000 to 100,000 lb).

Most of these were Earth launches of small orbit transfer vehicles. Those greater in mass than the currently planned shuttle capability of 29 500 kg (65,000 lb) can be launched partially off-loaded and subsequently topped off in orbit; this was the mode assumed in this study. It can be observed that potential benefits are likely if the shuttle were to be upgraded in up payload capability in the future; it is premature to state a specific desired capability.

Occasional needs were also seen for down payload capability greater than the 14 500 kg (32,000 lb) currently planned. These needs were principally for return to Earth of experiment or applications systems modules up to 21 000 kg (46,000 lb) in mass. The penalty for not leaving this capability is loss of opportunity to return systems to Earth for reuse; mission capability is not impacted. Again, it is premature to state a firm requirement as the above values are preliminary.

2.3.3.2 Heavy Lift

The Task 1 activity found potential heavy lift requirements rangine from 53 000 kg (117,000 lb) to 109 000 kg (240,000 lb) in a low Earth orbit. Task 2 results indicate the following potential requirements and considerations:

- It may be desirable to size the Heavy Lift Launch Vehicle (HLLV) to launch common stages fully fueled, leading to a mass range from 115 000 kg (250,000 lb) for LO_2/LH_2 common stages to possibly as much as 190 000 kg (420,000 lb) for LO_2/MMH common stages.

- It appears unnecessary to size the HLLV for the large single stages, up to 300 000 kg (660,000 lb). It may be more practical to launch these partially fueled and complete fueling on subsequent flights.

- Tanker flights fueling directly from residual propellant in the shuttle ET, used as a heavy lift element, may be practical. If no payload is flown, the ET arrives in orbit with residual propellant nearly equal to the payload that would otherwise have been delivered. This tanker mode (applicable only to LO_2/LH_2 orbit-to-orbit systems) would avoid development of a separate tanker system.

- The HLLV may be expected to fly several times a year. Accordingly its high-cost elements should be recovered and reused.

A heavy lift option that meets the above requirements was shown in Figure 2-3. This option was not compared to alternatives and does not represent a recommendation. The heavy lift requirements for power satellites (low cost heavy lift) are unique and are discussed in paragraph 2.3.5.

2.3.4 Orbit-to-Orbit Transportation

Several alternatives were investigated. These included a range of sizes, staging options, and propellants. Significant commonality between orbit-to-orbit transportation requirements exists among manned lunar and geosynchronous orbit missions. Systems sized for these missions can also accommodate the automated planetary options. The mass commonality is evident in Figure 2-9, showing principal LO_2/LH_2 options. The smallest vehicles are suitable for lunar landing when configured as landers and can be staged and augmented with expendable tanks to meet the larger orbit-to-orbit requirements. The largest vehicles provide single-stage reusable round trip capability. The intermediate size results principally from the common stage "slingshot" mode. In this mode, the booster stage propels the assembly into an elliptic orbit, reserving enough propellant for its own

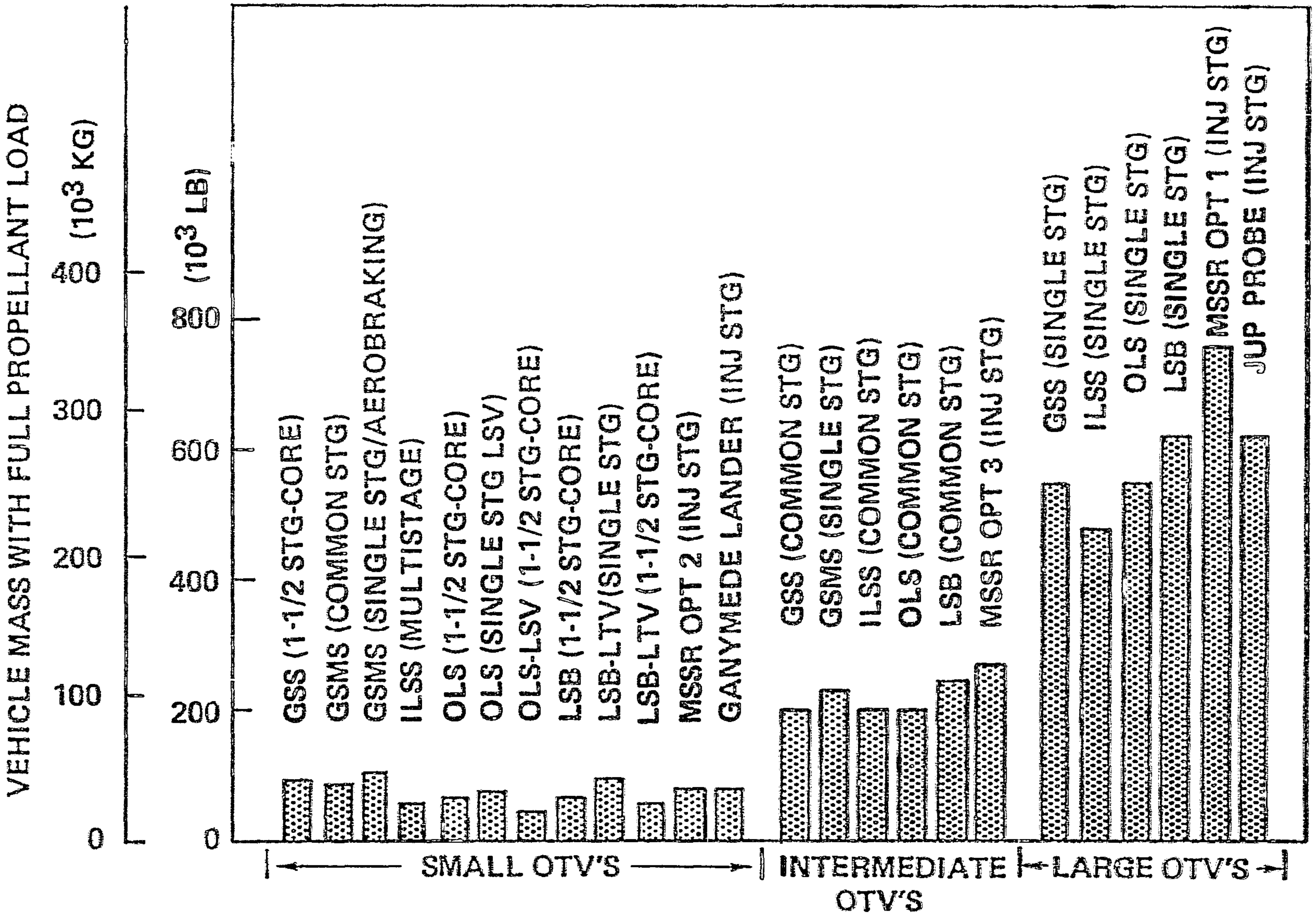


Figure 2-9. Mass Comparison of LO_2/LH_2 Stage Options

return to low circular orbit for recovery. The upper stage goes on from the ellipse to complete the mission, including return of payload and recovery of itself into a low circular orbit.

All modes investigated are compared for the manned geosynchronous missions in Figure 2-10 as to relative masses and in Table 2-3 as to other factors. The nuclear electric tug has less resupply (propellant) mass than any other system, but its problems with slow transfer (140 days up), and reactor disposal after its nuclear fuel is spent, may be adequate grounds for rejection. In other applications (see power satellite discussion below) its mass efficiency may be grounds for selection despite the problems. Solar-thermal electric propulsion systems (STEPS) offer a possible alternative to the nuclear tug. Performance is expected to be comparable, and no reactor radiation or disposal problems are encountered. Conceptual analyses of STEPS are included in the appendix to this report. It was not the intent of the effort reported here to select systems. Synthesis and selection of optimal systems will be accomplished in the planned Phase II of the present study.

Selection of the data presented as a comparison among systems was predicated on an evaluation of many alternative options for accomplishment of mission requirements. Figure 2-11 compares the options for the geosynchronous manned station mission in terms of parametric sizing of a common-stage LO_2/LH_2 OTV system. It is clear that the two-part delivery options (delivery of the station from low orbit to geosynchronous orbit) compare most nearlv with the crew rotation/ resupply requirement and that the latter is slightly more difficult. Therefore vehicles were sized for crew rotation/resupplyy as the basis of the above comparisons.

A comparison similar to that of Figure 2-9, but for LO_2/MMH stages, is presented in Figure 2-12. No single-stage options are shown since the predicted propellant fractions and specific impulse for LO_2/MMH systems result in very marginal single-stage systems for these missions. Figure 2-13 illustrates relative sizes of typical orbit transfer vehicles.

The orbit transfer systems as presently understood may be summarized as follows:

- The nuclear systems appear to present operational and safety problems that may outweigh their mass efficiency in these applications. Final judgment is reserved for a later phase of study.

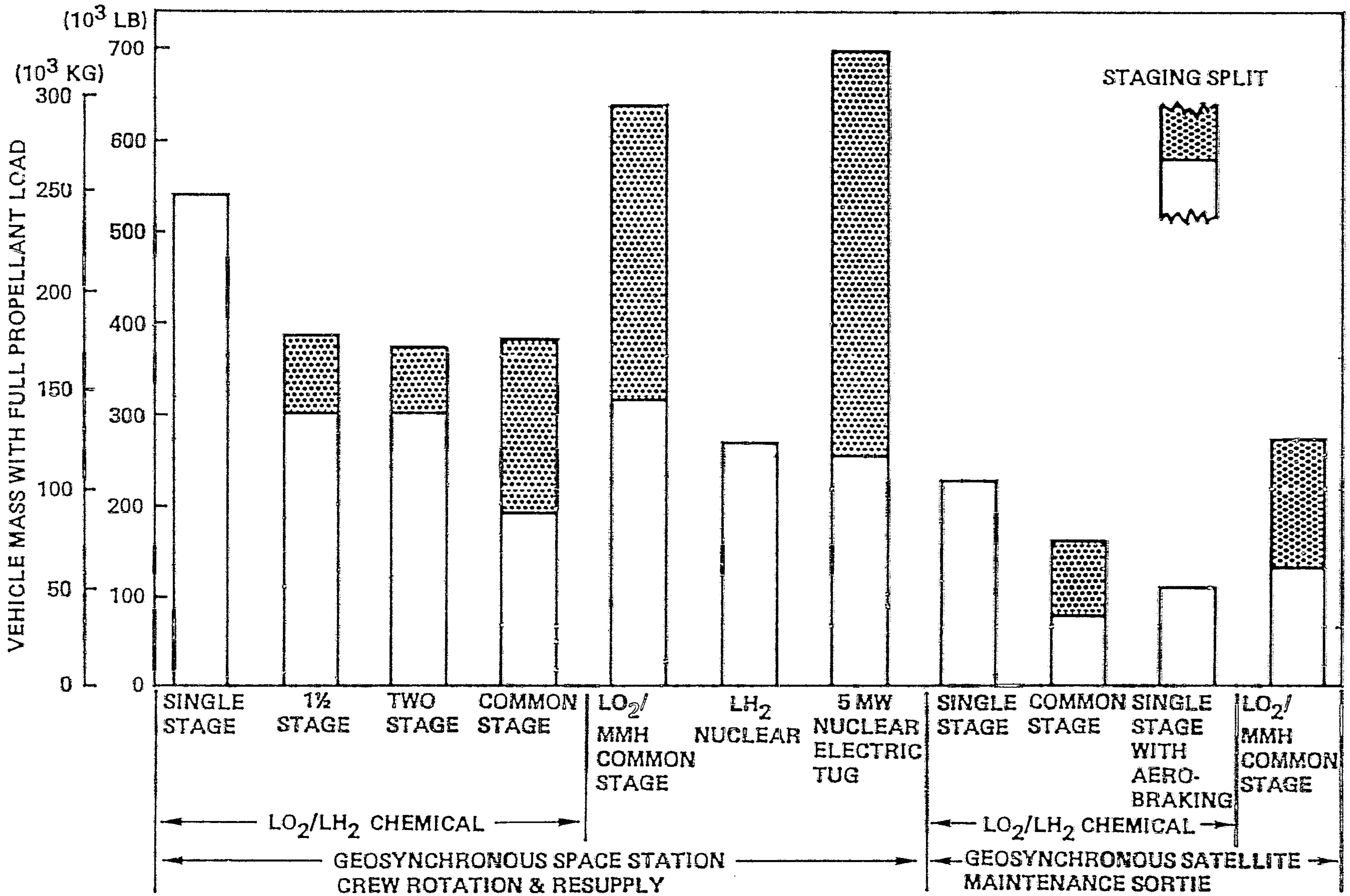


Figure 2-10. Geosynchronous Mission OTV Mass Comparison

Table 2-3. OTV Concept Assessment

OTV CONCEPTS*	COMPATIBLE WITH SHUTTLE	FULLY REUSABLE	NO. OF STAGES TO DEVELOP	MASS COMPARISON **	VOLUME COMPARISON**	ASSEMBLY REQ'D FOR REUSE	OPERATIONAL COMPLEXITY
SINGLE STAGE	NO	YES	1	1	1	NO	—
1-½ STAGE	YES	CORE	1 + DROP TANKS	.73	.65	YES	—
TWO STAGE	NO	RETURN STAGE	2	.68	.82	YES	—
COMMON STAGE	NO	YES	1	.70	.73	YES	• MONITOR 2 STAGES
COMMON STAGE (LO_2/MMH)	YES	YES	1	1.18	.43	YES	• MONITOR 2 STAGES
NUCLEAR STAGE (LH_2)	NO	YES	1	.47	1.90	NO	• RADIATION • REACTOR DISPOSAL
NUCLEAR ELECTRIC	NO	YES	1	N/A	N/A	NO	• REACTOR DISPOSAL
SINGLE STAGE + AUTOBRAKING	NO	YES	1	0.5	.43	YES	• MULTIPASS THRU VAN ALLEN BELTS

*LO_2/LH_2 UNLESS NOTED
**BASED ON GSS CREW ROTATION/RESUPPLY FLIGHT

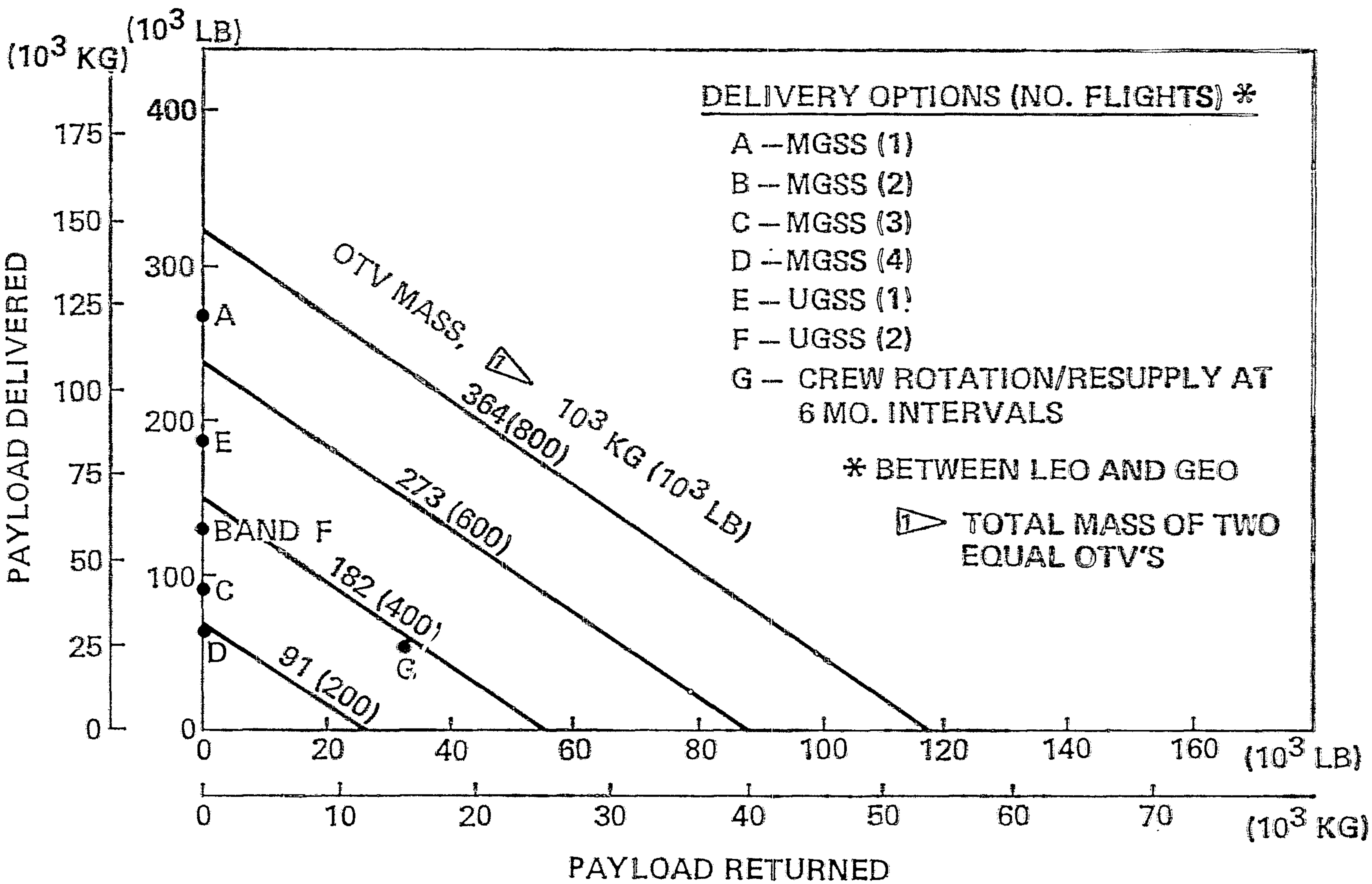


Figure 2-11. Common Stage LO_2/LH_2 OTV Capability for GSS

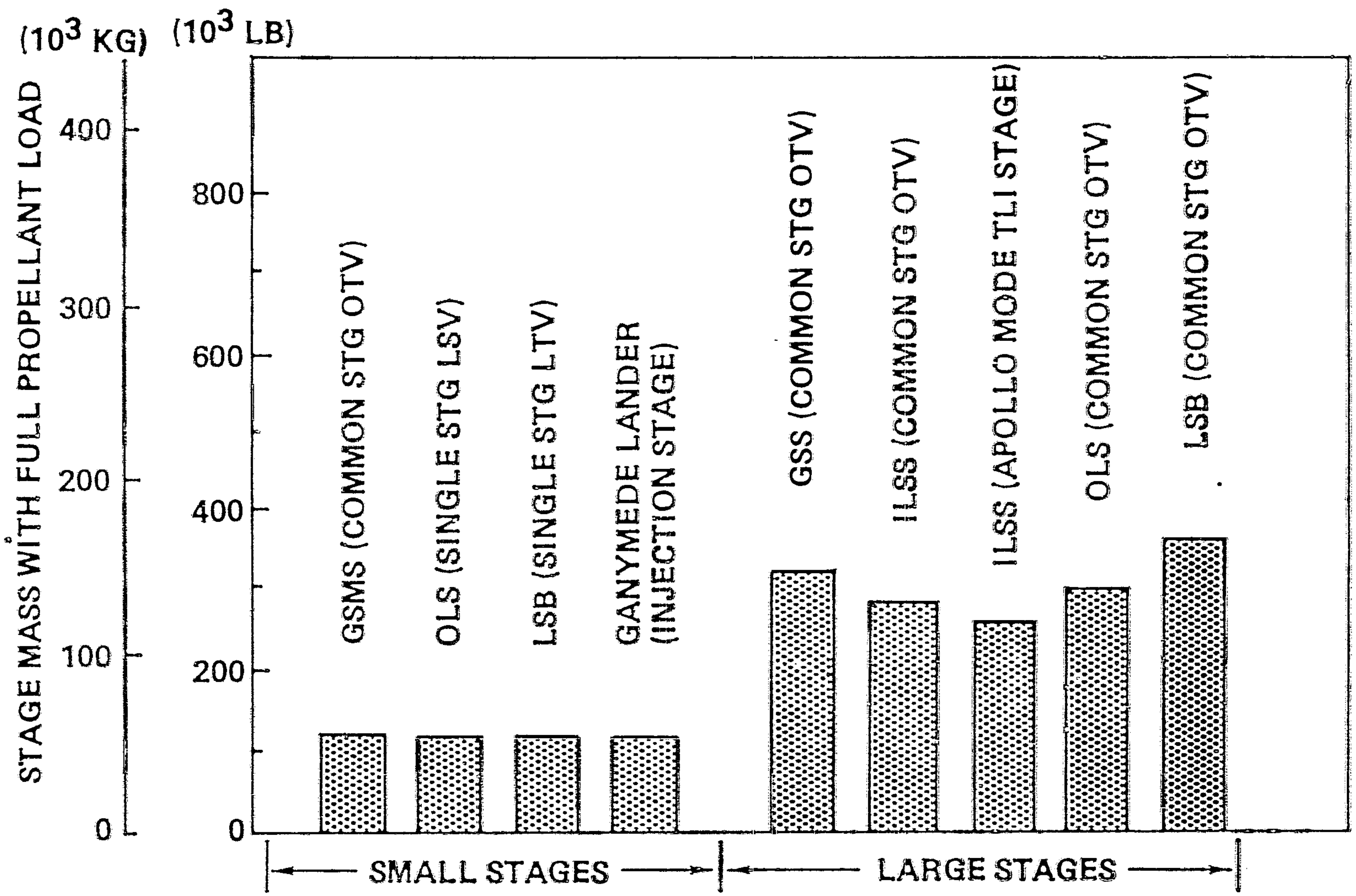
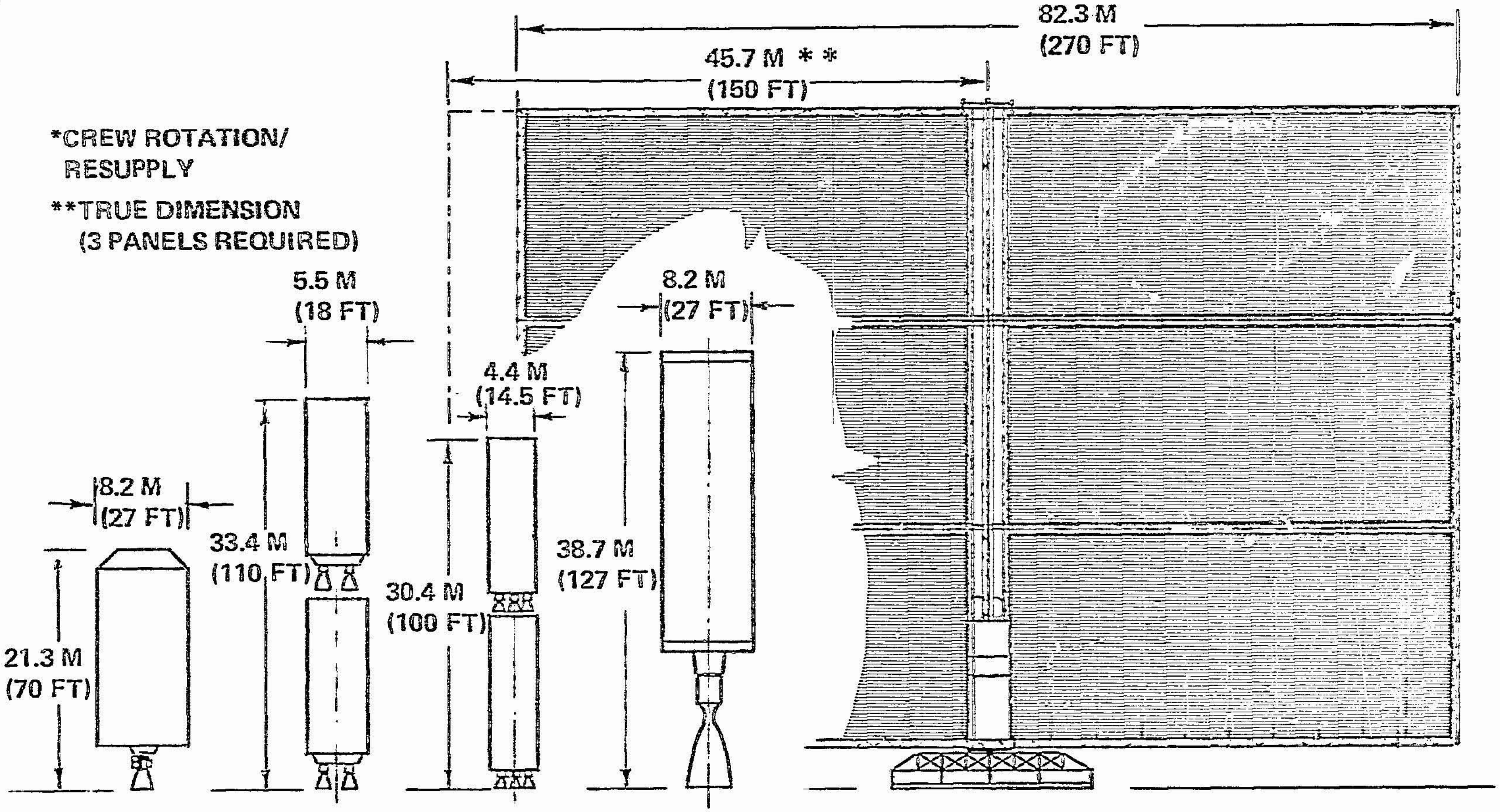


Figure 2-12. Mass Comparison of LO_2/MMH Stage Options



• SINGLE STG LO_2/LH_2 • 250 000 KG (550,000 LB)	• COMMON STG LO_2/LH_2 • 175 000 KG (385,000 LB)	• COMMON STG LO_2/MMH • 295 000 KG (650,000 LB)	• SINGLE STG NUCLEAR LH_2 • 127 270 KG (280,000 LB)	• SINGLE STG NUC ELECTRIC • 318 000 KG (700,000 LB)

*Figure 2-13. Representative OTV's for GSS Mission**

- The single-stage vehicle offers the simplest operations, but is not particularly efficient. Once placed on orbit it cannot be returned but rather must be refueled and refurbished on orbit.

- The 1-1/2 stage, with suitable sets of drop tanks, is efficient and compatible with shuttle launch. Orbital assembly is, however, complex, involving both side-to-side and end-to-end docking.

- The two (unequal) stage system offers no advantages as compared to the common stage system and is not recommended for further consideration.

- The common stage system offers full reusability in an efficient mode with only one stage development. The LO_2/LH_2 stages are too large for the shuttle and once placed on orbit, they must be refueled and refurbished there. If a 70% mass penalty is accepted, a common stage system with dense propellants (LO_2/MMH) can be defined having empty stages compatible with shuttle return to Earth.

Several aerobraking modes were investigated for the manned geosynchronous satellite maintenance missions; these have potential application to other manned geosynchronous missions. These modes influence sizing of the orbit transfer systems. The modes were:

- Use of an aerobraking (high drag device) kit on an OTV to allow circularization of the elliptic return transfer orbit by multiple aerobraking passes through the upper atmosphere rather than by propulsion.

- Use of an Apollo-type crew recovery vehicle. This vehicle is capable of direct entry from geosynchronous or lunar apogee. Thus the propulsion system does not have to deliver the payload through the low Earth orbit insertion delta V, and a shuttle crew recovery flight is not needed.

- Use of a winged crew recovery vehicle. The winged vehicle may include enough internal propellant for a coplanar deorbit from geosynchronous orbit. An L/D of 1.8 can provide plane changes from 0^o to 30^o during six to ten aerobraking passes that also reduce the orbit apogee from 35 786 km (19,323 n.mi.) to about 500 km (270 n.mi.). The final orbit at 30^o inclination overflies the United States; a shuttle-type entry and horizontal landing completes the recovery without the necessity for a shuttle crew recovery flight. Use of multiple aerobraking passes allows the heat shield to be designed only for low orbit entry.

2.3.5 Lunar Transport Vehicles

The lunar transport vehicles required for lunar program options in this study are potentially size-compatible with shuttle as shown in Figure 2-14. These vehicles, however, with the exception of the LO_2/LH_2 single stage and LO_2/MMH options, are quite long and may be risky to land due to too high a center of gravity this is especially true of the two-stage option. A problem also arises as to cargo placement. The cargo may be as voluminous as the LTV; perhaps more so for the lunar surface base mission. Figure 2-15 shows typical cargo volumes for a lunar landing operation with payloads side-mounted to a single-stage LO_2/LH_2 LTV. Also shown is a 1-1/2 stage lander not constrained to shuttle dimensions illustrating the height reduction thereby obtained.

Potentially attractive landers include:

- The LO_2/MMH single-stage lander, because of its compactness, low c.g. and freedom from LH_2 boiloff concerns;
- The LO_2/LH_2 single-stage lander because of its weight advantages and potential commonality with OTV applications; and
- Tailored-configuration landers designed expressly for the landing/ascent operation, because of the low c.g. attainable and ability to maximize vehicle utility for the unique needs of lunar landing and ascent.

2.3.6 Unique Transportation Needs

Three missions were identified as having unique transportation needs. These were manned Mars landing, nuclear (total waste option) waste disposal, and power satellites.

Manned Mars Landing--The degree of uniqueness of this requirement is associated with the critical issue of mission duration. In any event, a unique Mars landing/ascent vehicle is required. The ascent element may have a degree of commonality with the small orbit transfer/lunar lander stages previously described, but the landing element is a unique aerodynamic and rocket braking system.

Mars roundtrip stopover missions come in two principal varieties, a "short", typical 450-day, opposition-class mission, and a "long", typically 1,000-day, conjunction-class mission. Intermediate-duration (600-700 days) opportunities employing Venus swingbys also occur, somewhat less frequently. Energy (delta V) requirements are inversely correlated with duration. The nominal long mission has an orbit-to-orbit delta V requirement about 50% greater than that for Earth-moon missions. With elliptic orbits at Mars and direct entry Earth return, the requirement may be even

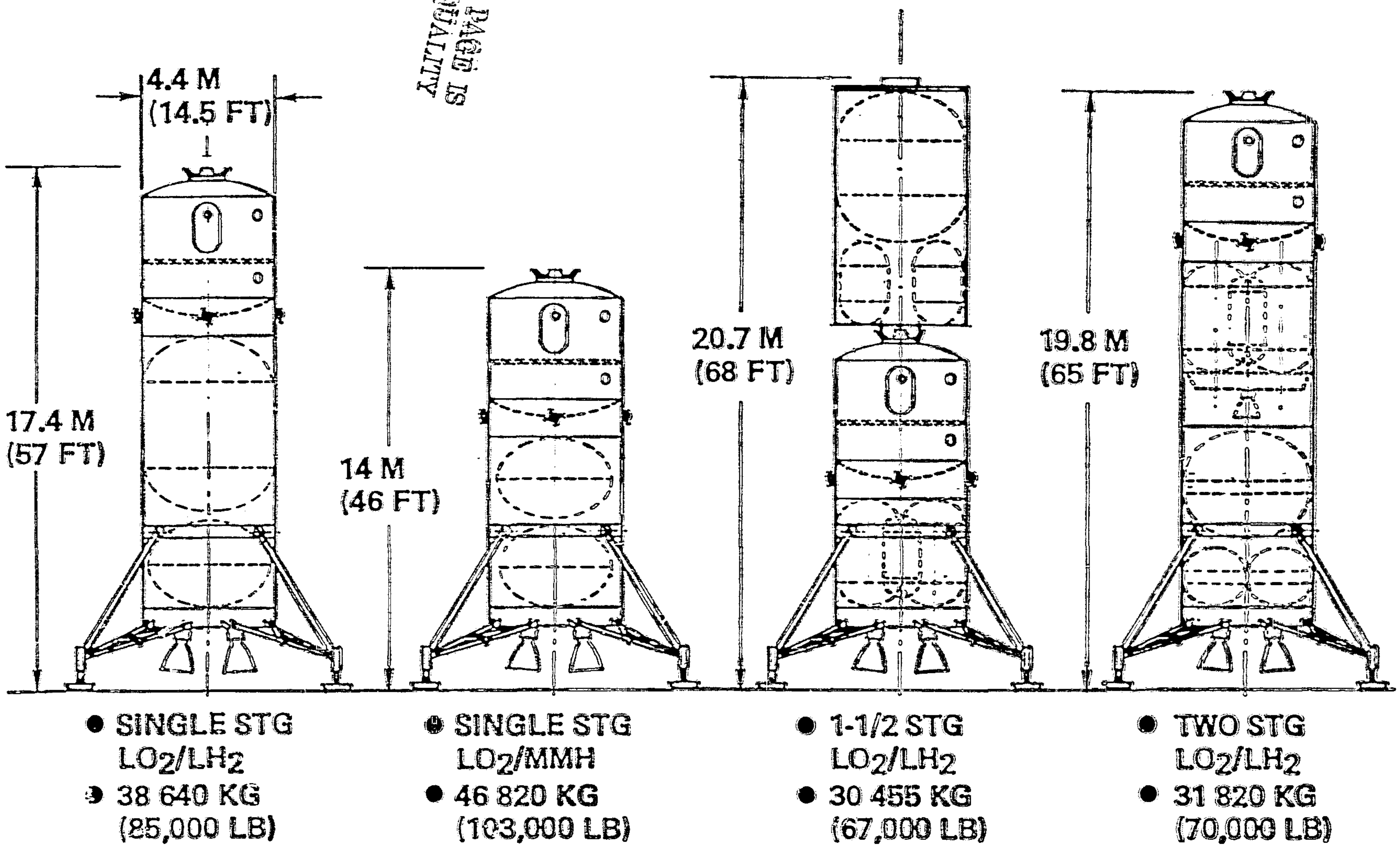


Figure 2-14. Representative LSV's for OLS Mission

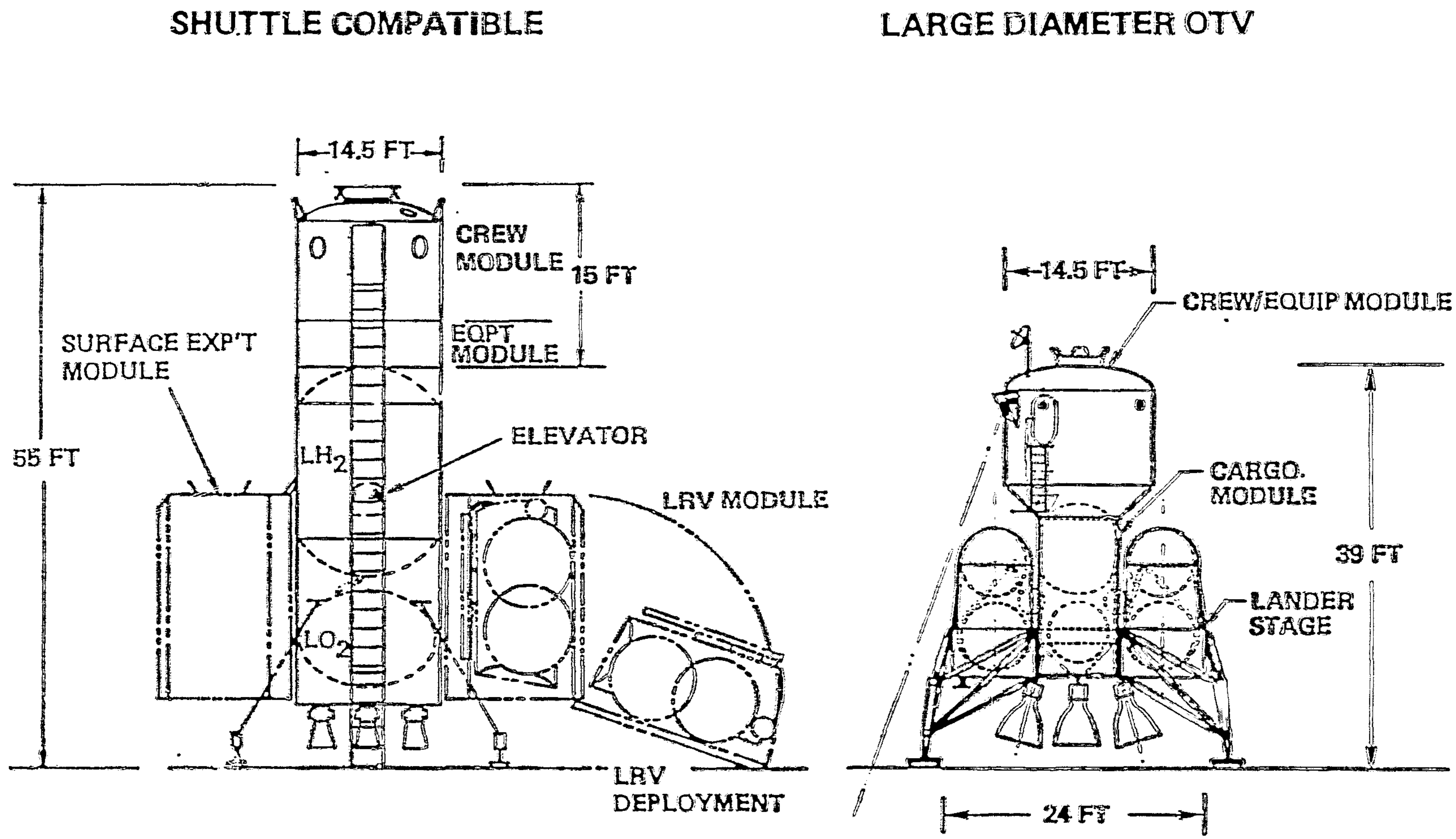


Figure 2-15. Small OTV Configurations — Lunar Landing/Ascent Application

less than the nominal Earth-moon roundtrip delta V. The nominal short mission has a delta V requirement about 100% greater than that for Earth-moon requirements, even with direct entry at Earth return. The Venus swingby missions are intermediate in energy requirements. The payload requirements (122 000 kg; 270,000 lb trans-Mars and 60 000 kg; 150,000 lb return) result in the following general situation:

- The long mission is compatible with the LO_2/LH_2 orbit-to-orbit stages described for geosynchronous and lunar missions, with a moderate amount of clustering. Less than 10 heavy-lift flights would be needed to assemble the system in Earth orbit.

- The short mission with LO_2/LH_2 requires either a great amount of clustering or development of a unique, very large Earth departure stage, and over 20 nominal heavy lift flights.

- Venus swingby combined with aerodynamic braking for Mars arrival results in an intermediate duration mission comparable masswise to the long mission, with LO_2/LH_2 propulsion. The Earth return stages plus the mission spacecraft must be enclosed in a large aerobraking heat shield through Mars arrival. This heat shield is again a unique requirement.

- LH_2 nuclear propulsion (Nerva-type hydrogen heater reactor) results in a 40%-50% reduction of the initial mass in Earth orbit as compared to LO_2/LH_2 propulsion for the short mission.

- A large nuclear-electric tug, such as described below for power satellite transfers to geosynchronous orbit, could provide a manned Mars round trip in 400 to 500 days, expending roughly 500 000 kg (1,100,000 lb) of propellant. The tug mass is roughly 450 000 kg (1,000,000 lb). The tug would return itself and the mission systems to Earth orbit for reuse.

Nuclear (Total Waste Option) Waste Disposal—The requirement imposed by this program option, as characterized by Task 1, was to launch, annually to solar system escape, 1100 waste packages each 4 500 kg(10,000 lb) mass. This is an average rate of 3 packages per day. This requirement, it should be pointed out, is not the baseline nuclear waste disposal mission (see Section 3.9) below but is the extreme case of disposal of total solidified waste.

It was found that potentially economical solutions to this requirement necessitated development of totally reusable systems. Costs of disposal in terms of kilowatt-hours of electricity generated were still about an order of magnitude greater than for the baseline case of disposal of refined wastes using the Shuttle. The total waste problem is described in more detail in the Appendix to this report.

Power Satellites—These systems are to be placed in geosynchronous orbit in order to collect and convert solar power and transmit it to Earth by microwave beam for commercial and industrial use. Like the nuclear (total waste option) waste disposal mission, the transportation requirement is large and must meet economics constraints resulting in a requirement for totally reusable low-cost transportation.

Various studies and analyses of power satellites have been reported. Estimates of specific mass have ranged from 2.3 kg/kw (5 lb/kw) to 6.5 kg/kw (14 lb/kw), with power delivered to the grid per satellite from 5,000 to 10,000 megawatts. The range of specific mass results primarily from various assumptions as to the degree of technology advance, over today's state of the art, represented by the satellite.

Solar photovoltaic and solar concentrator thermal engines have been proposed. Thermal concentrators apparently can operate in the high-intensity regions of the van Allen radiation belts with possible minor degradation of collector reflectivity, whereas solar photovoltaics will suffer significant degradation of output unless the solar cells are well shielded or unless major strides are made in reducing solar cell sensitivity to radiation. Thus it is likely that the thermal engine plants could generate power to propel themselves (by electric propulsion) to a geosynchronous orbit, whereas the photovoltaic plants are expected to require an orbit transfer system such as a nuclear-electric tug.

A solar photovoltaic satellite delivering 5,000 megawatts power (on the ground) at a total mass of 11.3×10^6 kg (25×10^6 lb) as defined in NASA CR-2357 is used as an example for this summary. A nuclear electric tug is assumed for orbit-to-orbit transfer. The tug was sized such that the reactor core, neutron reflector, and inner gamma shield can be returned to Earth as a unit of 28 000 kg (63,500 lb) mass by the shuttle, for refueling. The inner gamma shield is presumed to provide, after a cooldown period, enough shielding of the spent core to protect shuttle and ground crews. The sizing rationale described results in a tug with 20 megawatts jet power, and a total mass of 475 000 kg (1,050,000 lb). The tug can deliver about 550 000 kg (1,000,000 lb) to a geosynchronous orbit in 160 days, using a total of 360 000 kg (800,000 lb) of propellant, including that needed for tug return. A total of 25 tug flights will deliver the entire satellite to geosynchronous orbit. A fleet of 25 tugs is required to keep the total delivery time within one year. The total mass to low Earth orbit, excluding the tugs themselves and other logistics, is 20.4×10^6 kg (45×10^6 lb) to place each satellite in its geosynchronous orbit.

The key parameters in economics feasibility of power satellites are:

- The system development investment required
- The cost and weight of the satellites and their ground-based power receiving antennas
- The value of power derived from the system, i.e. competitive busbar cost
- The cost of ground and space operations associated with launch, assembly, operations, and maintenance of the systems
- The cost of transportation from low Earth orbit to geosynchronous orbit, and most significant, the efficiency of that transfer as reflected into total mass transportation requirements to Earth orbit
- The cost of transportation to low Earth orbit

Various economics studies have projected a competitive busbar cost at about $0.025/kwh in 1975 dollars. This value, together with preliminary estimates of development and operations costs, and assumed use of electric propulsion for the geosynchronous transfer, results in the tradeoff of satellite cost and weight, and low orbit transportation cost, shown in Figure 2-16. Observations that may be derived from the figure are that:

- The satellites must be producible, including costs of orbital assembly, at costs per unit weight (i.e. dollars per pound), comparable to those experienced for commercial or military jet aircraft. Since the satellites will be large and simple structures compared to aircraft, this appears feasible provided that orbital assembly costs can be kept within reasonable bounds.
- Cost of low orbit transportation must be in the range $20 – $100/kg ($10 – $45/lb). Payload per flight should be large to aid in minimizing orbital assembly costs. Preliminary studies show that these costs and characteristics are attainable by a next-generation launch vehicle incorporating total reusability.
- Economic feasibility of a total satellite energy system presents challenges to technology and systems development but on the basis of available data appears within reach.

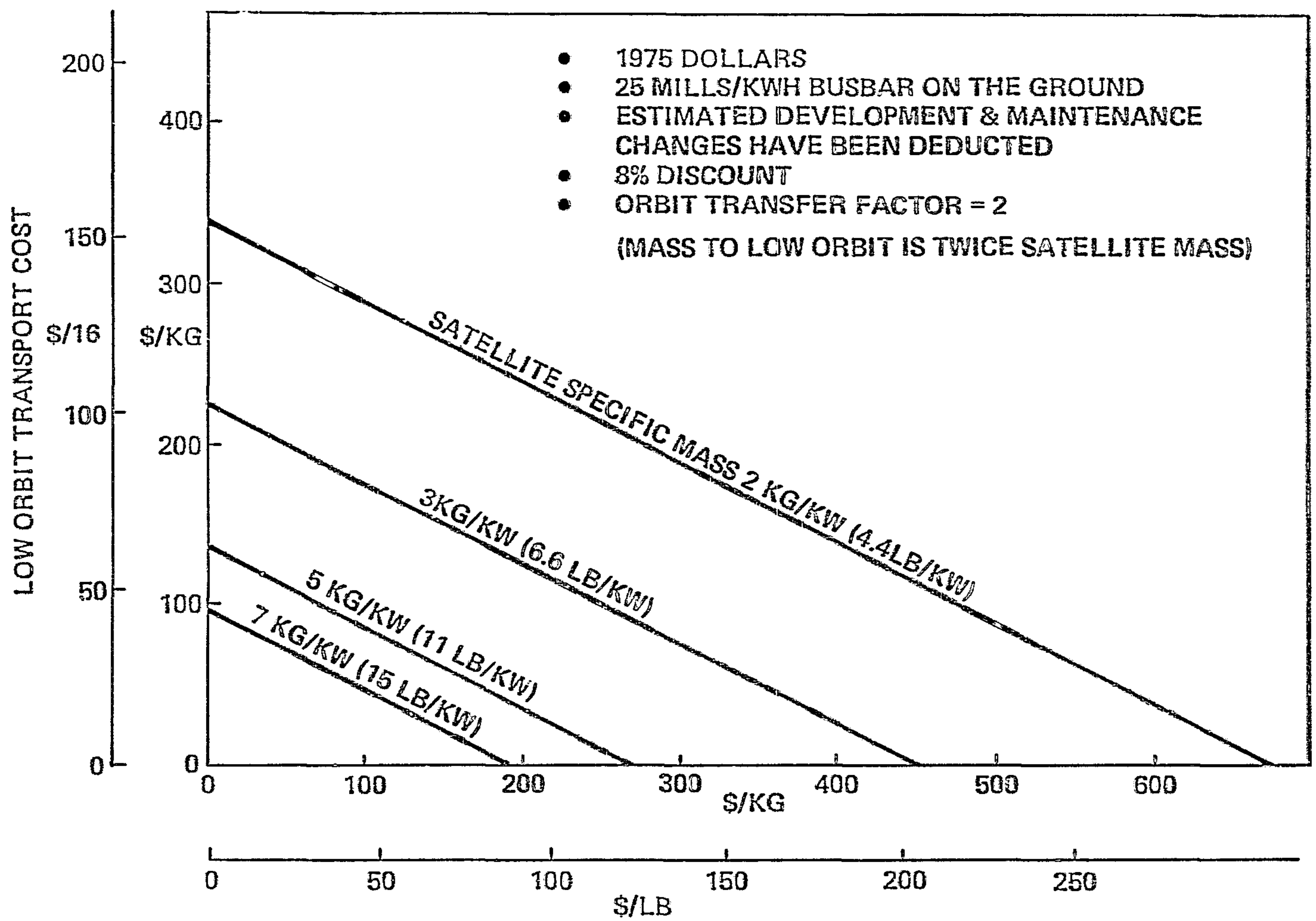


Figure 2-16. Power Satellite Economics Overview

3.0 SPACE PROGRAM OPTIONS AND MISSION MODES ANALYSES

Each program option is individually reported in the following pages. It is assumed that each program option is independent of the others.

Mass requirements are treated in the following way:

- Mass properties reported under mission system descriptions reflect identified mass properties without growth allowances.

- Growth allowances are included in statements of transportation requirements. Growth allowances were determined on the basis of a statistical study of historical mass growth (summarized in the appendix). The allowances applied are commensurate with a 50% statistical confidence they would not be exceeded.

Technology improvements were examined briefly. Two improvements were found to be of some significance to the major manned station missions--solar arrays and integrated avionics. The total weight savings constitute a small percentage of the payload system mass; significant changes in transportation requirements were not found.

3.1 EARTH ORBIT SPACE STATIONS PROGRAM

Three options for manned stations in a low Earth orbit were examined. The modular space station illustrated in figure 3.1-1 can be delivered to orbit in modules by the space shuttle and assembled in orbit; it does not require advanced space transportation. The unitary station is a single large-core module with attached applications and science modules (ASM's) tailored to specific missions. The unitary station is too large for the shuttle and must be placed in orbit by a heavy-lift vehicle. Both of these stations normally carry a 12-man crew. The third option is the space base, built up from unitary station sized modules in a manner analogous to the assembly of smaller modules. The space base will support a crew of up to 60 men and can be expanded beyond that if desired. All of these stations nominally operate at 500 km (270 nmi) altitude and 55 degree inclination, but other inclinations might be used for some applications. The principal references were the Rockwell space station studies during the period 1970-1972.

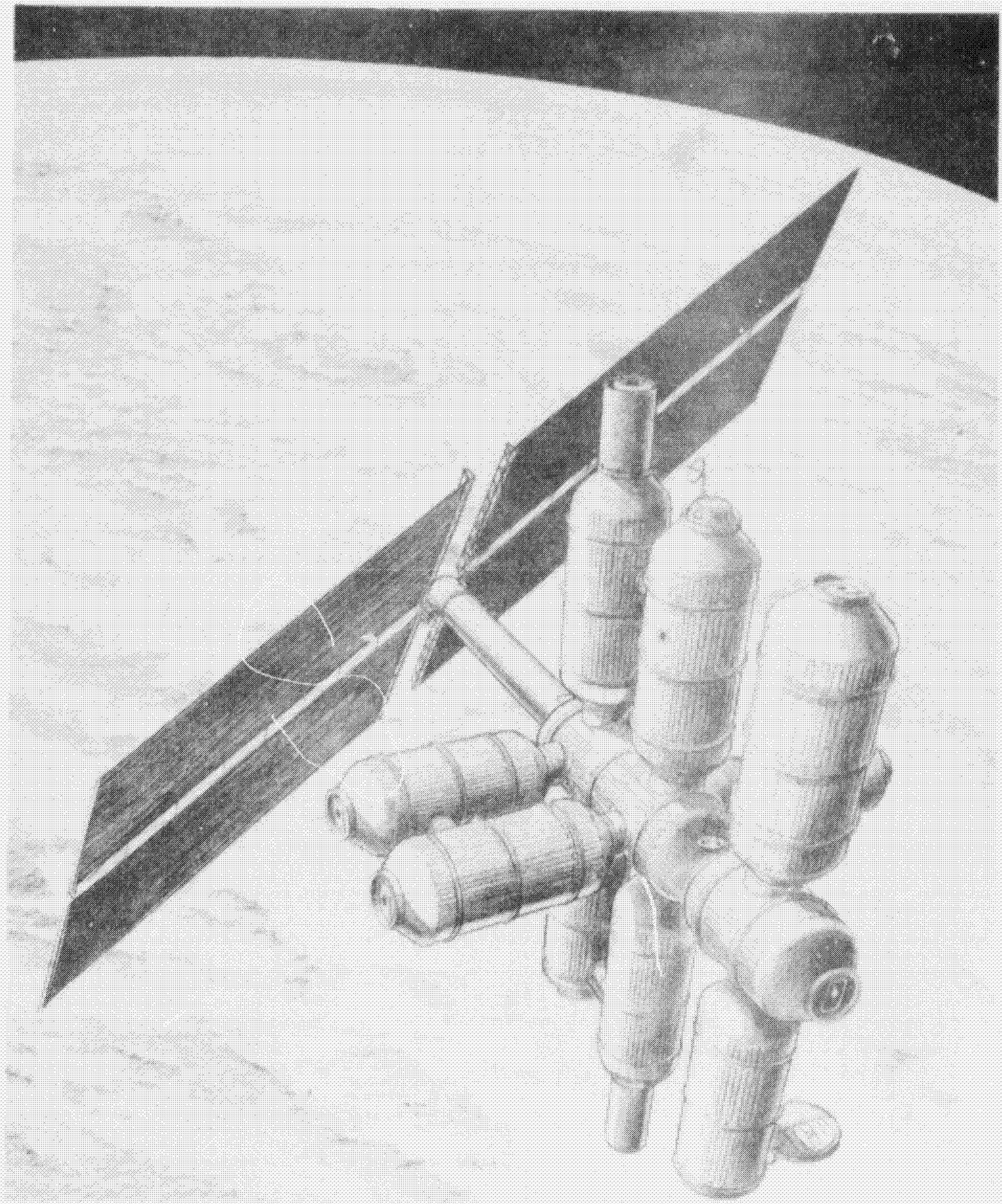


Figure 3.1-1: Modular Earth Orbiting Space Station

3.1.1 TWELVE-MAN EARTH ORBIT SPACE STATIONS

3.1.1.1 Mission Summary

3.1.1.1.1 General Description

These missions include modular and unitary (single-module) station options for support of up to twelve men in a low Earth orbit for a variety of potential science and applications missions. The modular station modules are defined such that stations for 3, 6, 9, or 12 men can be built-up from the module set. Only the 12-man configuration is described here.

3.1.1.1.2 Mission Assumptions and Constraints

General mission assumptions and constraints are summarized in table 3.1-1.

3.1.1.2 Mission Systems Descriptions

3.1.1.2.1 Mission Options

3.1.1.2.1.1 Modular Earth Orbit Space Station

The nominal flight configuration for the MSS is shown in figure 3.1-2. Nine station modules are required to provide quarters for the 12-man crew, supporting subsystems, and consumables. Four docking ports are available for attached and detached application and science modules (ASM's).

No crew transfer vehicles are shown at the station since the crew is transported between Earth and the station in a crew transfer module (CTM) that is located in the cargo bay of the space shuttle orbiter. Resupply modules (RM's) are used to house crew consumables and station expendables; these modules are also transported by the space shuttle. Following integration of a new RM into the station system, the expended RM is returned to Earth for refurbishment and subsequent reuse on another resupply mission.

The orbital weight of the MSS with two ASM's and a full RM is approximately 98 600 kg (217,000 lb) as identified by the reference.

3.1.1.2.1.2 Unitary Earth Orbit Space Station

The flight configuration for the 12-man unitary Earth orbit space station (USS) is shown in figure 3.1-3. The USS differs from the modular space station in that a single large module provides the housing for the crew, general-purpose labs, and subsystems.

Table 3.1-1: Low Earth Orbit Space Station Assumptions and Constraints

MISSION	OBJECTIVES	MISSION ASSUMPTIONS & CONSTRAINTS

MANNED STATION IN EARTH ORBIT	• LONG TERM MANNED SPACE RESIDENCE
	• SCIENTIFIC INVESTIGATION OF THE NEAR EARTH SPACE
	• BROAD SPECTRUM OF EARTH OBSERVATION
	• ADAPTABILITY TO OBSERVATION OF OTHER CELESTIAL OBJECTS (SOLAR, STELLAR, ETC.)
	• SPACE MANUFACTURING TECHNOLOGY DEVELOPMENT
		• INTERMEDIATE TO HIGH INCLINATION EARTH ORBIT (BASELINE 55°, 500 KM. RANGE 28.5 – 97°, 400-600 KM)
		• COMPATIBLE WITH SPACE SHUTTLE FOR LAUNCH (MODULAR ONLY) AND RESUPPLY
		• UNITARY STATION PLACED ON ORBIT IN SINGLE LAUNCH
		• DOCKING SYSTEM FOR:
		  • ADDITIONAL STATION MODULES (UP TO TBD)
		  • LOGISTIC (RESUPPLY) MODULES
		  • SPACELAB EXPERIMENT MODULES
		  • FREE-FLYING MODULES
		• BASIC CREW SIZE OF SIX, TO 12; UP TO 60 FOR SPACE BASE
		• UNMANNED QUIESCENT (STAND-DOWN) BY GROUND CONTROL
		• CONTROLLED DEORBIT AT END OF LIFE
		• CREW EVA CAPABILITY

ASM's are attached to the station as well as a resupply module (RM). As in the modular space station, no crew transfer vehicle remains at the station since the crew can be rescued within 1 day by an Earth-launched space shuttle.

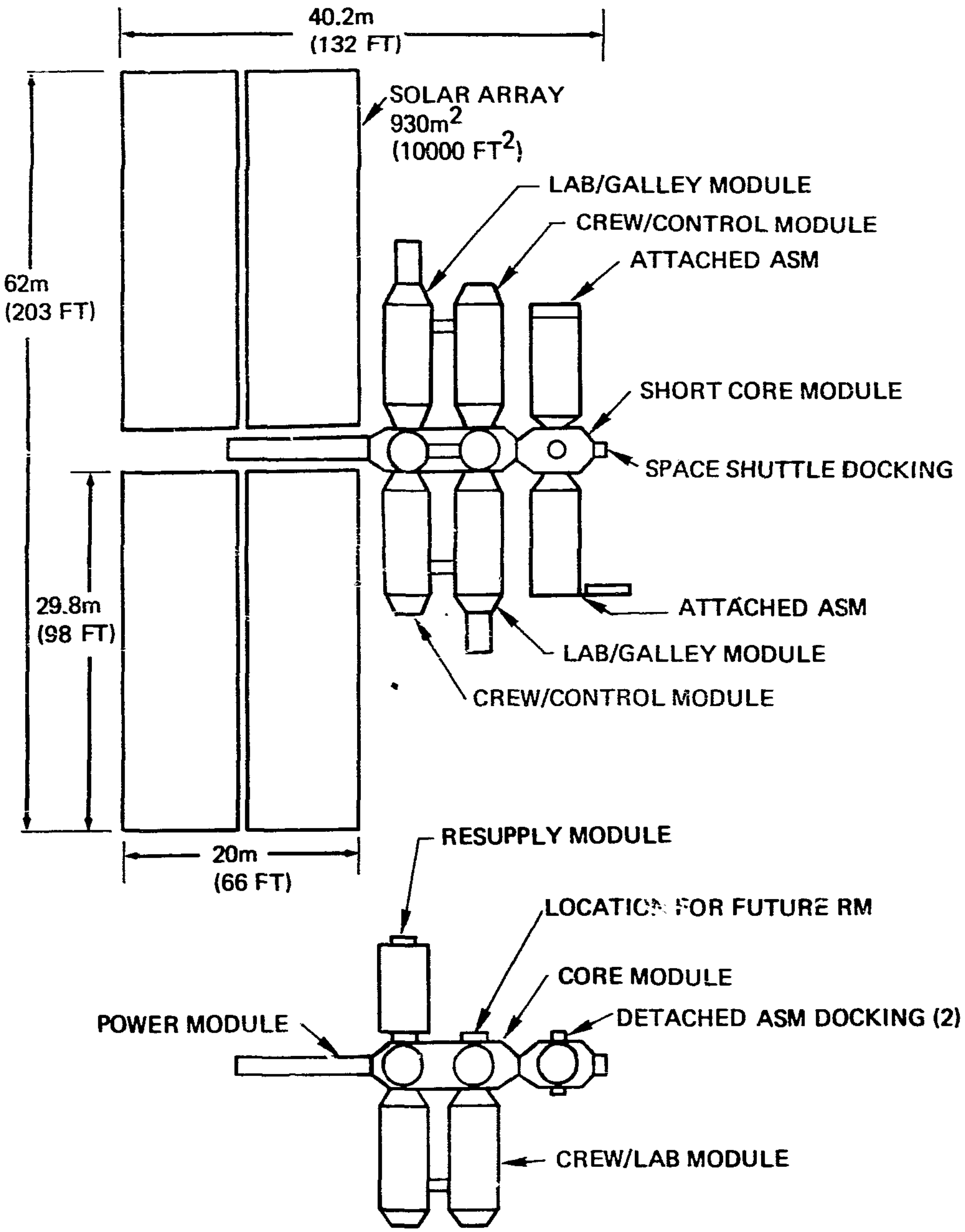
The total orbital mass with two ASM's and a RM is 81 820 kg (180,000 lb) as identified by the reference.

3.1.1.2.2 Payload Description

3.1.1.2.2.1 Modular Earth Orbit Space Station Payloads

A brief description of size and weights of the nine modules forming the station is presented in table 3.1-2.

The basic core module (CM) houses basic station subsystems such as G&N and RCS and provides the framework to allow attachment of many of the other modules. The short core module (SCM) is used primarily to provide docking facilities for the RAM's. The power module (PM) supports the 930-m^2 (10,000-ft^2) solar array that provides the average load of approximately 25 kw. Also included is cryogenic storage for fuel cells and batteries.



STATION CHARACTERISTICS

BASIC MODULES	9
RESUPPLY MODULE	1
ASM	2
BASIC STATION MASS	75,000 KG (165,000 LBS)
TOTAL STATION MASS	97,700 KG (215,000 LBS)

NADIR

Figure 3.1-2: 12-Man Modular EOSS

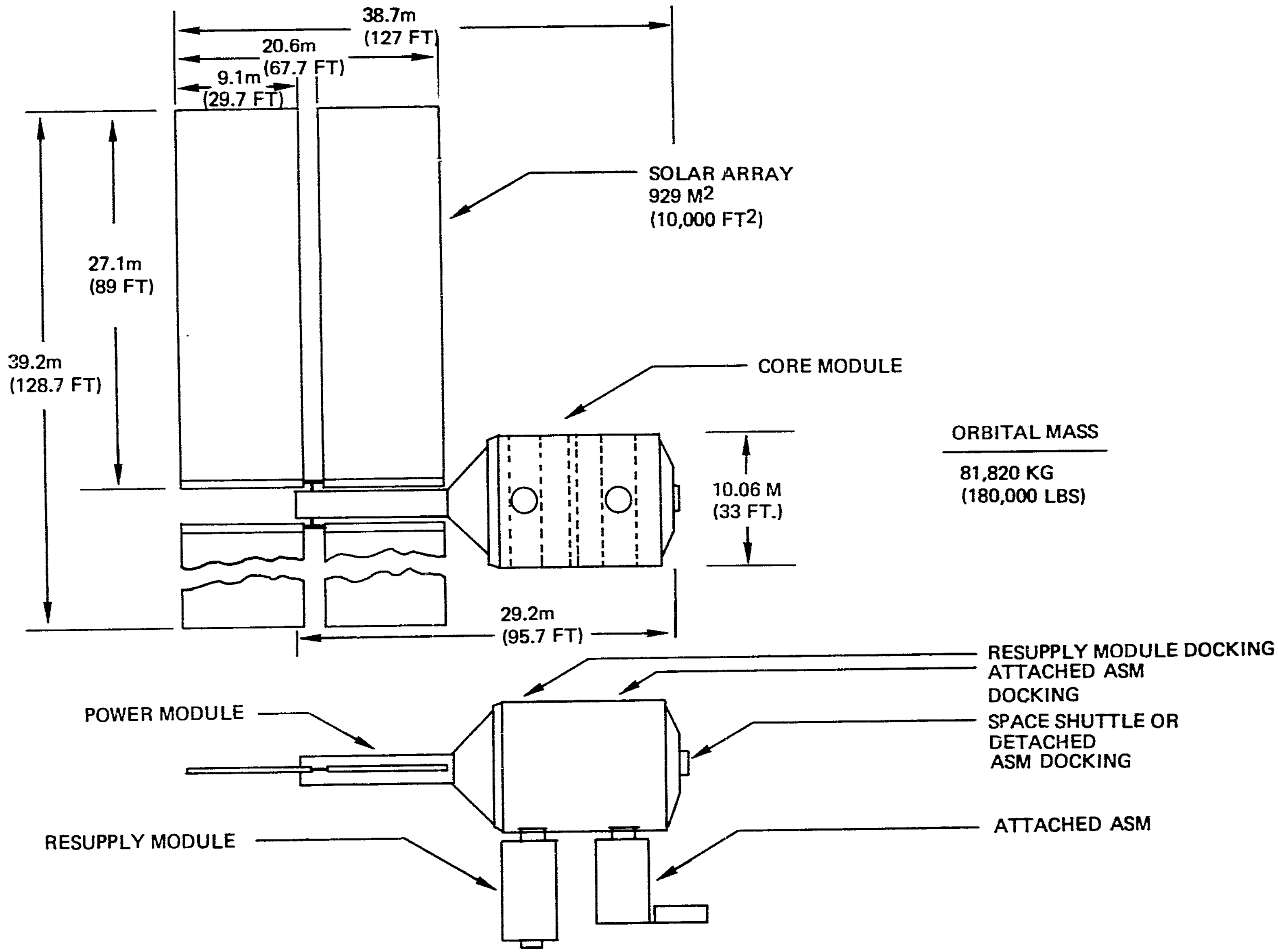


Figure 3.1-3: 12-Man Unitary EOSS

Table 3.1-2. MSS Basic Modules

ITEM	DESCRIPTION	QTY	UNIT WEIGHT		UNIT SIZE (D x L)	
			10^3 KG	10^3 LBS	M	
1. CORE MODULE (CM)	ATTACHMENT FOR ALL OTHER MODULES BASIC STATION SUB SYS (GN, RCS)	1	10	22	4.4 x 12.2	14.5 x 40
2. POWER MODULE (PM)	SOLAR ARRAY 930m^2 HIGH PRESS. GAS STORAGE	1	7	15.5	4.4 x 10.7	14.5 x 35
3. CREW/CONTROL MODULE (CCM)	STATEROOMS (3 EACH) ECLSS FOR 6 (EA) CONTROL CENTER/MEDICAL PERSONNEL HYGIENE	2	9.1	20	4.4 x 11.8	14.5 x 38.7
4. LAB/GALLEY MODULE (LGM)	GENERAL PURPOSE LAB GALLEY & DINING/RECREATION. EVA AIRLOCKS	2	8.5	18.7	4.4 x 11.8	14.5 x 38.7
5. SHORT CORE (SC)	ATTACHMENT FOR RAMS	1	4.5	10	4.4 x 6.1	14.5 x 20
6. CREW/LAB MODULE (CLM)	STATEROOMS (3 EA) ECLSS FOR 6 (EA) GEN. PURPOSE LAB	2	9.1	20	4.4 x 11.8	14.5 x 38.7
	TOTAL		74.9	164.9		

Each of the two crew/control modules (CCM) provides crew quarters and station control capability. Each of these modules has three staterooms and environmental control equipment suitable to satisfy the needs of six men. Each of the two crew/lab modules (CLM) also provide crew quarters for three as well as general-purpose lab facilities. Each module has environmental control systems sized for six men. The two lab/galley modules (LGM) each provide galley, dining and recreation facilities and general-purpose labs.

All of the modules are 4.3m (14 ft) diameter and vary in length between 6.1m (20 ft) and 12.1m (40 ft). The mass of the modules varies from 4 545 kg (10,000 lb) to 10 000 kg (22,000 lb). The center of gravity for each module is estimated to be at its geometric centroid.

3.1.1.2.2.2 Unitary Earth Orbit Space Station Payloads

The basic unitary station for the Earth orbit mission includes a core module and power module.

The core module provides quarters for the 12-man crew and houses the majority of the subsystems. The module is divided into two separate pressure compartments for safety reasons. Each compartment consists of two transverse decks. One of the pressure compartments includes a deck for experiments and another serves as a combination crew quarters and station control deck. The second pressure compartment also includes a crew/control deck and a deck to provide galley, dining, recreation, and medical facilities. Toroidal end bulkheads provide additional volume for housing subsystems and equipment. The core module has a diameter of 10.06m (33 ft) and length of approximately 18.6m (61 ft).

The power module supports the 930-m^2 (10,000-ft^2) solar array and houses the secondary and emergency power systems. The module has a length of 11.7m (38 ft).

The dry mass of the basic station, including the integral experiments and effective mass of the launch fairing, is approximately 40 635 kg (109,200 lb) as indicated by the reference study. The initial consumables launched with the station supply 12 men for 3 months and total 4 860 kg (10,700 lb). The center of gravity of the station is estimated to be roughly 2m (6.6 ft) forward of the geometric centroid of the core module.

3.1.1.2.2.3 Application and Science Module (ASM) Payloads

The bulk of the applications and science program defined for the 12-man low-Earth-orbit space station is accomplished through use of application and science modules (ASM's). These modules provide the means to conduct science and application missions, basic engineering, and advanced subsystem operations. ASM central control as well as biomedical research will be accomplished

within the lab modules of the station.

The ASM's defined herein are typical of transportation requirements but do not necessarily represent current planning or concepts for actual programs that would be carried out by manned Earth orbit space stations.

Attached ASM's—Seven types of ASM's were defined in the reference as having operating characteristics compatible with the station or requiring crew involvement. These were operated in attached mode. At any one time, not more than two of these ASM's would be present. Two of the normally attached ASM's as originally defined (see table 3.1-3) are larger than the cargo bay of the orbiter and therefore would be redefined or eliminated if transportation is confined to the space shuttle.

Table 3.1-3. Attached Application and Science Modules

ITEM	DESCRIPTION	UNIT WEIGHT		UNIT SIZE (DXL)	
		10^3 KG	10^3 LBS	M	FT
1. HIGH ENERGY STELLAR GAMMA RAY MODULE	HIGH ENERGY SURVEY-GAMMA RAYS	5.64	12.4	4.3 x 8.1	14 x 26.5
2. EARTH SURVEY MODULE	EARTH SURVEYS	7.18	15.8	4.3 x 8.1	14 x 26.5
3. MANNED CENTRIFUGE MODULE	CENTRIFUGE	2.14	4.7	4.3 x 5.6	14 x 18.5
4. ISOTOPE-BRAYTON POWER MODULE	ISOTOPE-BRAYTON TEST SYSTEM	9.55	21.0	4.3 x 16.6	14 x 54.5
5. PRIMATE	PRIMATE PHYSIOLOGY AND BEHAVIOR	1.36	3.0	4.3 x 4.3	14 x 14
6. SPACE BIOLOGY MODULE	SMALL VERTEBRATES SPECIMENS MICROBIOLOGY INVERTEBRATES	12.4	27.2	6.7 x 11.9	22 x 39

Detached ASM's—These ASM's are generally those that have attitude stability and pointing requirements that exceed the capability provided by the station. Several detached ASM's may be in orbit at one time and each will be moved to the station for refurbishment at approximately 90-day intervals. The characteristics of the detached ASM's are presented in table 3.1-4.

It should also be noted that the majority of the detached ASM's include a basic instrument/sensor package plus a support section that provides the subsystems to operate and control the instruments/sensors. The combined length of the two units in many cases exceeds the space shuttle's cargo-bay length. Two launches are required to get these detached ASM's into orbit.

The totals as tabulated are for the modular station option. The unitary station option as described in the reference required more experiment resupply, due apparently to a higher level of experiment activity. Accordingly, the resupply interval was six weeks rather than eight weeks. It should be recognized that resupply requirements and interval are highly dependent on the particular science and applications programs in process and may be expected to vary significantly over the life of a station program.

Table 3.1-4. Detached ASM's and Support Sections

ITEM	DESCRIPTION	UNIT WEIGHT		UNIT SIZE (DXL)	
		10^3 KG	10^3 LBS	M	FT
1. X-RAY ASTRONOMY MODULE	GRAZING INCIDENCE TELESCOPE, HIGH ENERGY SURVEY	7.54	16.6	4.3 x 13	14 x 42.5
	SUPPORT SECTION	5.23	11.5	4.3 x 7.6	14 x 24.5
2. ADVANCED STELLAR ASTRONOMY MODULE	ADV. STELLAR TELESCOPE (3M)	8.72	19.2	4.3 x 17.8	14 x 58.5
	SUPPORT SECTION	5.23	11.5	4.3 x 7.6	14 x 24.5
3. UV SOLAR ASTRONOMY MODULE	1.5 DIFFU[illegible] LIMITED TELESCO[illegible] EXTREME UV SPECTR[illegible] HELIOGRAPH	7.77	17.1	4.3 x 17.8	14 x 58.5
	SUPPORT SECTION	5.23	11.5	4.3 x 7.6	14 x 24.5
4. SOLAR CORONOGRAPH & X-RAY MODULE	1-6, 5-30 SOLOR CORONOGRAPH. X-RAY TELESCOPE	4.5	9.9	4.3 x 13	14 x 42.5
	SUPPORT SECTION	5.23	11.5	4.3 x 7.6	14 x 24.5
5. UV STELLAR PLATFORM MODULE	UV STELLAR PLATFORM	1.5	3.3	2.2 x 3.9	7 x 8.5
	SUPPORT SECTION	5.23	11.5	4.3 x 7.6	14 x 24.5
6. UV SCHMIDT AST. MODULE	0.6 UV SCHMIDT TELESCOPE	4.82	10.6	4.3 x 13	14 x 42.5
	SUPPORT SECTION	5.23	11.5	4.3 x 7.6	14 x 24.5
7. REMOTE MANEUVERING SUBSATELLITE		N/A	N/A	1.5 x 1.2	5 x 4
8. COSMIC RAY LAB MODULE	COSMIC RAY LAB	17.2	37.9	4.3 x 13	14 x 42.5
	SUPPORT SECTION	7.2	15.9	4.3 x 7.6	14 x 24.5
9. FLUID PHYSICS MODULE	FLUID PHYSICS	17.3	38.1	4.3 x 17.8	14 x 58.5
	SUPPORT SECTION	5.2	11.5	4.3 x 7.6	14 x 24.5
10. IR STELLAR AST. MODULE	IR STELLAR SURVEY	4.2	9.2	4.3 x 8.1	14 x 26.5
	SUPPORT SECTION	5.2	11.5	4.3 x 7.6	14 x 24.5

3.1.1.2.2.4 Crew Transfer and Resupply Payloads

Crewman stay times in the low orbit stations could presumably be as long as six months. The nominal resupply interval is two months, providing opportunities for more frequent crew rotation if that should be desired. The controlling requirement is resupply of experiments and consumables. Since the payload capability of the shuttle is greater than the resupply payload at the nominal two-month interval, longer intervals could be used. Additional flights will be required occasionally for ASM delivery or changeout. Some ASM's are small enough to be carried on a regular resupply flight; others will require dedicated flights.

Crew Transfer Module—Transportation for the crew between Earth and the station is provided by the space shuttle. However, since the orbiter crew compartment cannot accommodate 12 station crewmen and at least 3 orbiter crewmen, a separate module is used to house the station crewmen. This crew transfer module (CTM) will be carried in the cargo bay of the orbiter. The entire station crew is normally rotated at once; the CTM is therefore sized for 12 men. The CTM will also be self-sufficient in terms of providing environmental control/life-support subsystems. The CTM is not designed to be released from the orbiter while in orbit. Its estimated mass, excluding passengers, is 4 500 kg (10,000 lb).

Resupply Module (RM)—The RM is designed to have a pressurized section for the storage of bulk cargo (food, clothes, spares, etc.) and an unpressurized area for storage of fluids. The 2-month resupply interval specified in the modular station reference required a total of 5 000 kg (11,000 lb) of cargo.

The mass of the resupply module including unusable consumables but excluding usable consumables is estimated to be 4 800 kg (10,580 lb).

Consumables—A listing of the consumables and other resupply items included in the RM for a 2-month interval is shown in table 3.1-5.

3.1.1.2.2.5 Mass Summary

Mass summaries for the 12 man stations are summarized for mission start-up and resupply in tables 3.1-6 and 3.1-7 respectively.

These values are identified masses and do not include the growth allowance appropriate to sizing transportation systems.

Table 3.1-5. MSS Resupply Requirements (12 Men – 60 Days)

ITEM	MASS	
	KG	LB
CLOTHING	138	304
LINENS	112	248
GROOMING	18	40
MEDICAL	27	60
UTENSILS	102	224
FOOD	1179	2600
GASEOUS STORAGE		
OXYGEN	3	6
NITROGEN	342	754
WATER	650	1432
LiOH	9	20
WATER MANAGEMENT	73	162
ATMOSPHERE CONTROL	394	868
CO_2 MANAGEMENT	103	226
WASTE MANAGEMENT	48	106
HYGIENE	19	42
SPARES	63	138
SUBTOTAL	3280	7230
AV EXPERIMENT RESUPPLY	1633	3600
TOTAL 60 DAY AV.	4912	10,830
UP-DOWN EMERGENCY (96 HOURS)		
OXYGEN	287	633
NITROGEN	16	36
TOTAL LOADED CONSUMABLES	5216	11,499

Table 3.1-6. 12-Man Space Station Start-Up Mass Summary

Payload	Modular station		Unitary station	
	10^3KG	10^3LB	10^3KG	10^3LB
Basic station	74.9	165	49.6	109.3
Attached ASM's (typical)	12.7	30	12.7	30
Resupply module (RM)	10	22	10	22
Crew	1.1	2.4	1.1	2.4
Total	98.8	219.4	73.4	163.7

Table 3.1-7. 12-Man Space Station Resupply Mass Summary

Payload	Mass	
	10^3KG	10^3 LB
Delivery		
· CTM	4.5	10.0
· Crew	0.5	1.0
· RM	4.8	10.6
· Consumables	5.2	11.5
Total	15.0	33.2
Return		
· Delivery (less) consumables	(4.9)	(10.8)
· Plus science and crew effects	1.0	2.2
Total	11.1	24.6

3.1.1.2.2.6 Pickup Points and Transportation Constraints

Modular Station--These station modules include docking ports at either end that provide pickup points. For transportation in the shuttle payload bay, adapter fixtures will be required to bridge from docking points to payload bay attach points. These modules must be protected from aerodynamic loads during Earth launch.

Unitary Station--The unitary station incorporates structural hard points around its aft circumference for adaption to a heavy lift launch vehicle. The unitary station modules must be protected from aerodynamic loads during Earth launch.

ASM's, CTV's, and RM's--These are handled like the modular station modules.

3.1.1.2.3 Transfer and Storage

The RM's provide necessary storage and consumables. Transfer is not ordinarily required. Some ASM's are likely to require transfer of fluids from the RM.

3.1.1.2.4 Orbital Assembly, Maintenance, and Modification

The modular station is designed to be assembled in orbit by docking of modules. The unitary station is designed to be orbited in a single launch; assembly will be confined to attachment of ASM's. Routine maintenance will be provided in orbit by the station crew. Modular station elements can be returned to Earth by the shuttle if necessary; the unitary station does not permit this, but subsystems can presumably be modularized such that changeout of faulty hardware is possible with return to Earth for repair. Requirements for orbital modifications as such were not identified.

3.1.1.3 Transportation Requirements

3.1.1.3.1 Payload Delivery Points

Low Earth orbit is the delivery point for all payloads. The nominal orbit is at 55 degrees inclination; 500 km (270 nmi) altitude. Orbits from 28-1/2 degrees to sun synchronous and 300 km to 550 km (162 nmi to 297 nmi) are potential alternatives.

3.1.1.3.2 Payload Delivery Options

A mass growth allowance of 24% was used for transportation capability sizing. This allowance is applied to hardware but not to consumables.

3.1.1.3.2.1 Modular Station, CTM, RM, and ASM's

The modular station system includes only elements that can be delivered by the space shuttle.

Station modules are sized for delivery of one per shuttle flight. The CTM and RM normally make up a shuttle payload; small ASM's can be included on a CTM/RM flight. Large ASM's require dedicated flights.

The maximum station module mass, including growth, is 12 400 kg (27,340 lb). This is well within the shuttle delivery or return payload capability. The longest module is 11.8m (38.7 ft), permitting use of a shuttle OMS kit when needed. The nominal crew rotation/resupply payload is 17 200 kg (40,000 lb) delivered and 13 300 kg (29,400 lb) returned. The maximum ASM delivery mass is 21 500 kg (47,000 lb). This ASM mass exceeds the nominal shuttle return payload of 14 500 kg (32,000 lb).

3.1.1.3.2.2 Unitary Station

Crew transfer modules, resupply modules, and the majority of the ASM's are delivered by the space shuttle. The space station and the two large attached ASM's will require heavy-lift capability for delivery.

Some of the detached ASM's were too long for the orbiter's cargo bay and hence were designed to have the instrument/sensor unit separable from the subsystem support section. As a result, two launches were required to get a given ASM to orbit. With a heavy lift/large envelope transportation, these can be launched as a single unit.

The mass including growth of the basic station is approximately 66 410 kg (146,000 lb); it has an overall envelope of 10.06 mD x 30.2 mL (33 ft x 99 ft). The largest of the attached ASM's is the engineering and operations module that has a mass of 46 200 kg (101,000 lb) and an envelope of 7.6D x 20.17L meters (25 x 68 ft). Crew rotation/resupply payloads are the same as for modular station.

3.1.1.3.2.3 Operational Constraints

Normal shuttle operational procedures are applicable. HLLV launches for the unitary option will be unmanned.

3.1.1.4 Mission/Transportation Modes and Operations

3.1.1.4.1 Transportation Options

The space shuttle is the only applicable option for the modular station. The unitary station requires a HLLV for launch of the station and some of the large ASM's. The 2-SRB option of the SRB/ET HLLV as defined in paragraph 2.2.1.2 is adequate.

3.1.1.4.2 Representative Transportation Mode and Systems

3.1.1.4.2.1 Transportation Sequence

Mission operations involving transportation systems in the MSS program are shown in figure 3.1-4. The major operations include station build-up, application and science module build-up and crew rotation/resupply.

The major unitary space station program operations involving transportation systems are depicted in figure 3.1-5. In summary, the station is launched unmanned by a HLLV followed by the initial manning using the space shuttle. Delivery of ASM and periodic resupply and crew rotation is also performed with the space shuttle.

The shuttle trajectories will be standard. Typical HLLV trajectories are described in the appendix to this report.

3.1.1.4.2.2 Transportation Sizing

Not applicable.

3.1.1.4.2.3 Operational Factors

Mission Profiles, Timelines, and Constraints—Transfers to the station orbit by the shuttle or any other transportation vehicle are expected to pass through a 185 km (100 nmi) parking and phasing orbit. Direct ascents are comparatively inefficient. The entire ascent and rendezvous process normally requires only a few hours and is not a significant mission factor.

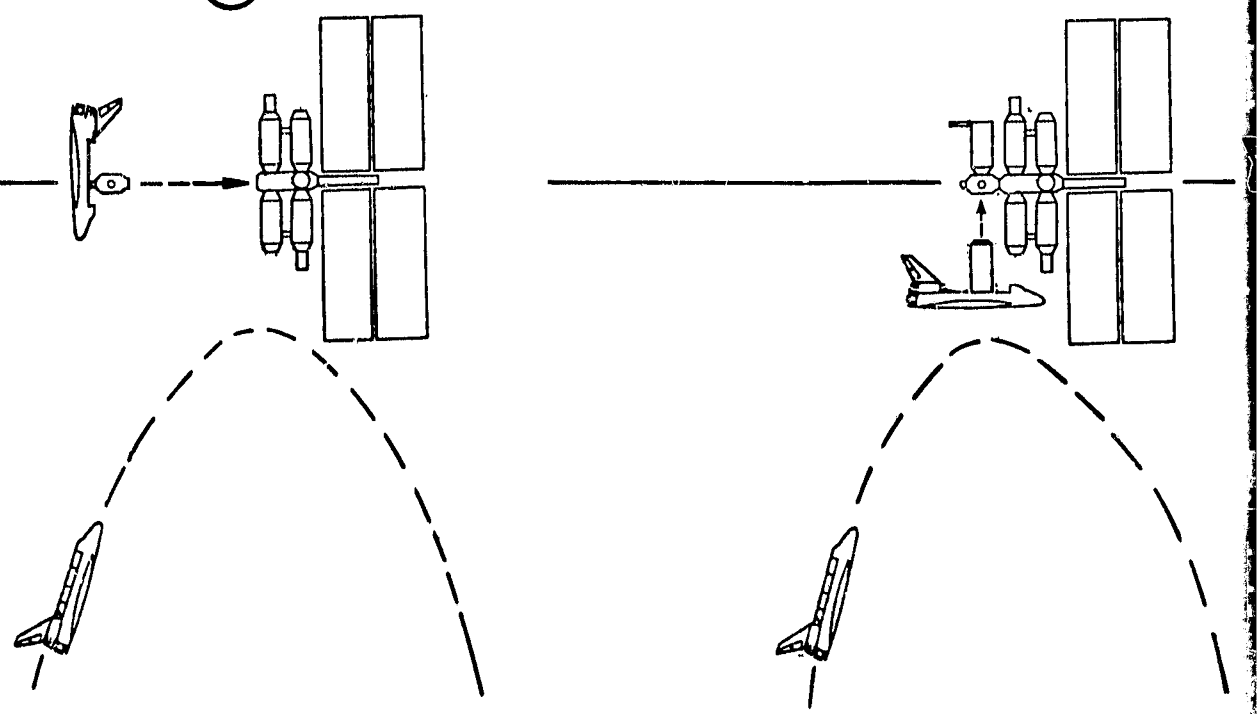
It is expected that repeating orbits would be selected for the space stations; for example, the 55°, 500-km (270-nmi) orbit results in 15 orbits per day. One ascent opportunity per day occurs with the station positioned in its orbit so that lengthy along-track phasing maneuvers are not required.

Space station missions are constrained in altitude by the van Allen belt radiation. The upper altitude limit is approximately 550 km (330 nmi), depending to some degree on orbit inclination and the amount of shielding afforded by the station.

Crew Involvement—All launches of the modular space station are made by the manned shuttle, so a flight crew is always present to assist docking and assembly operations during station build-up and subsequent operations. The unitary station is launched unmanned in a single launch by a heavy-lift vehicle. Subsequent launches (crew transfer, ASM's, and resupply) are made by the manned shuttle so that a flight crew is present to assist docking operations. The space base is assembled from large

STATION BUILD-UP	EXPERIMENT MODULE BUILD-U
1	2

EARTH ORBIT
270 N.M.
55 DEG.



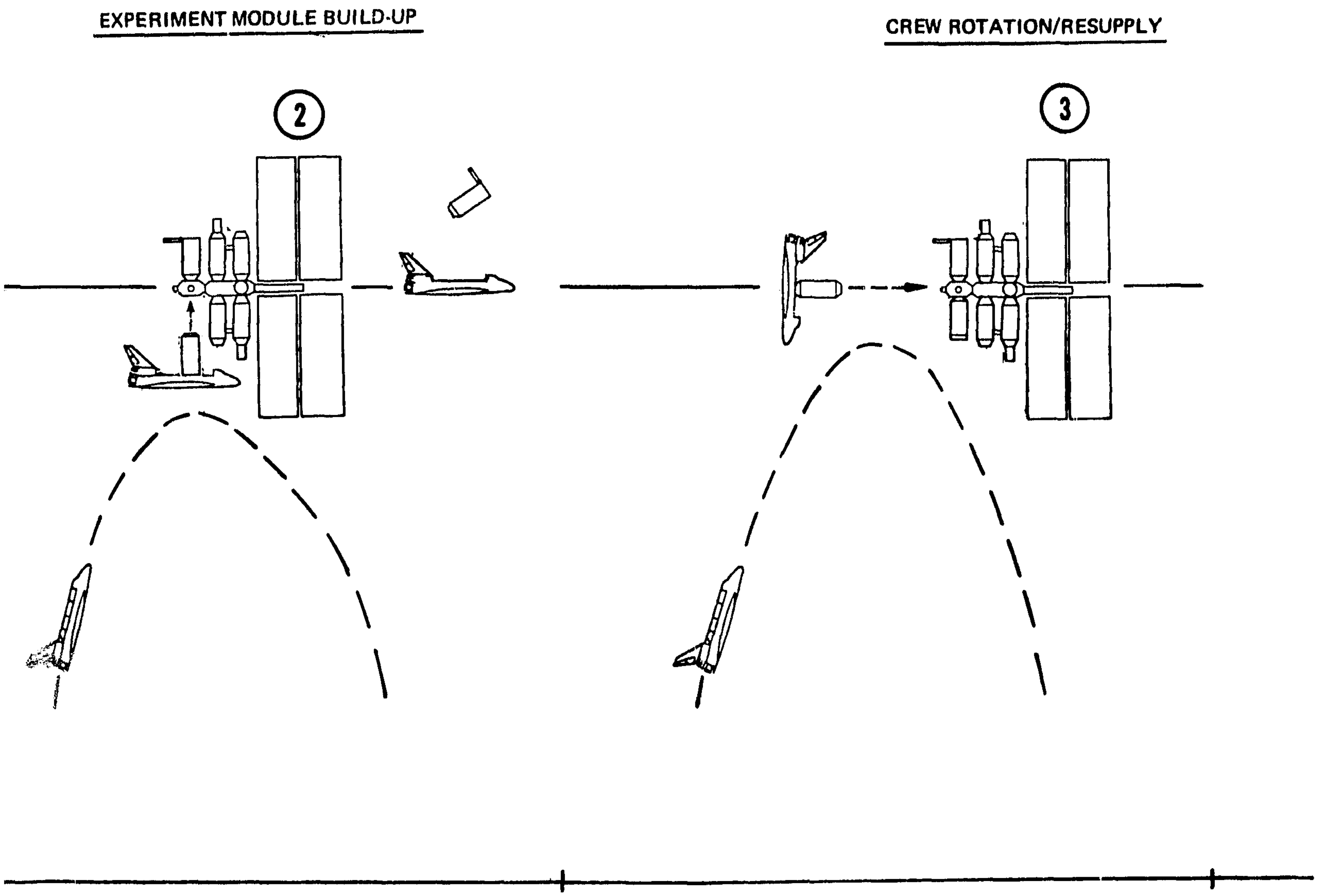
PHASE TIME — 9 WEEKS

CUM TIME

1	2
• DELIVER STATION MODULES AND 12 MAN CREW • 9 SS FLIGHTS	• DELIVER ATTACHED AND DETA EXPERIMENT MODULES • 4 SS FLIGHTS PER YEAR

FOLDOUT FRAME

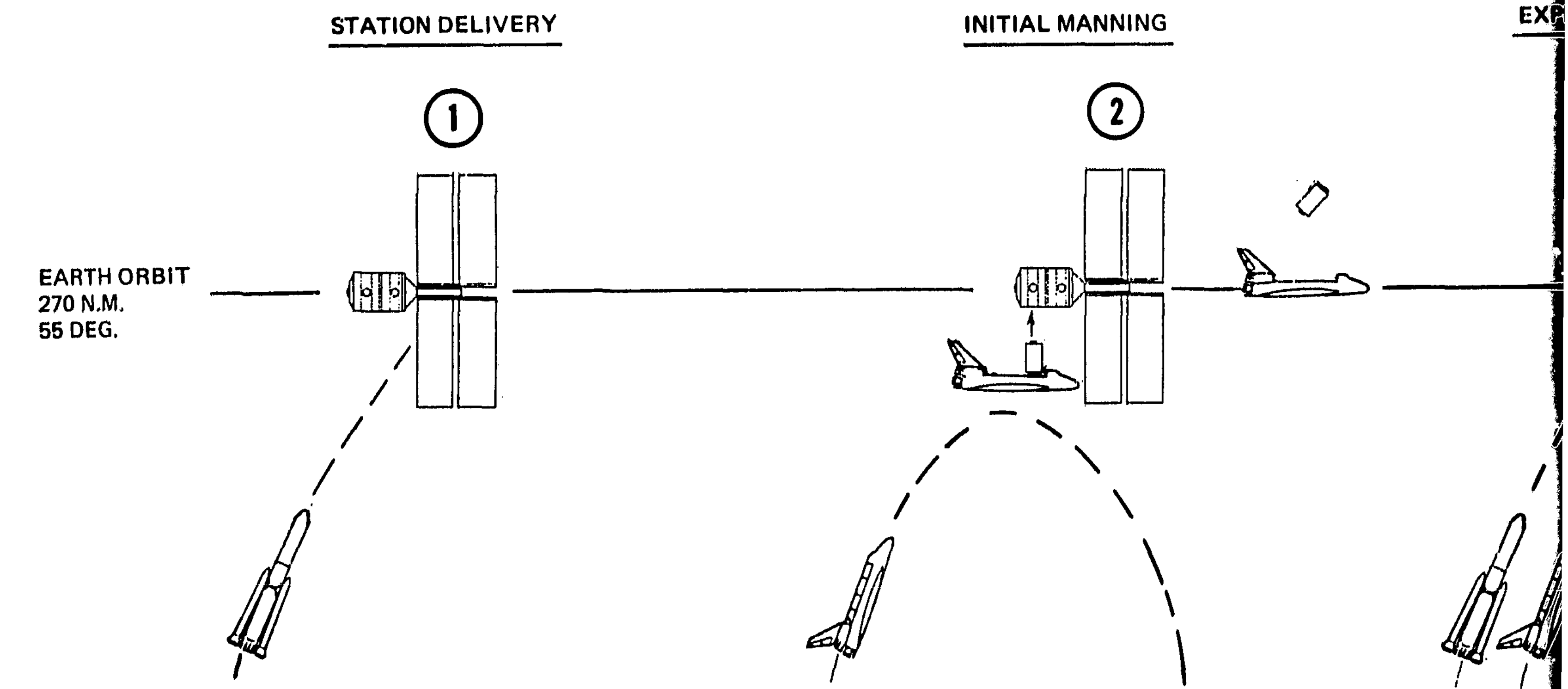
/



2	3
• DELIVER ATTACHED AND DETACHED EXPERIMENT MODULES • 4 SS FLIGHTS PER YEAR	• DELIVER RESUPPLY MODULE AT 3 MONTH INTERVALS • DELIVER CREW REPLACEMENTS ON EVERY SECOND RESUPPLY FLIGHT • TOTAL OF 4 SS FLIGHTS PER YEAR

Figure 3.1- 4 Modular Earth Orbit Space Station Sequence

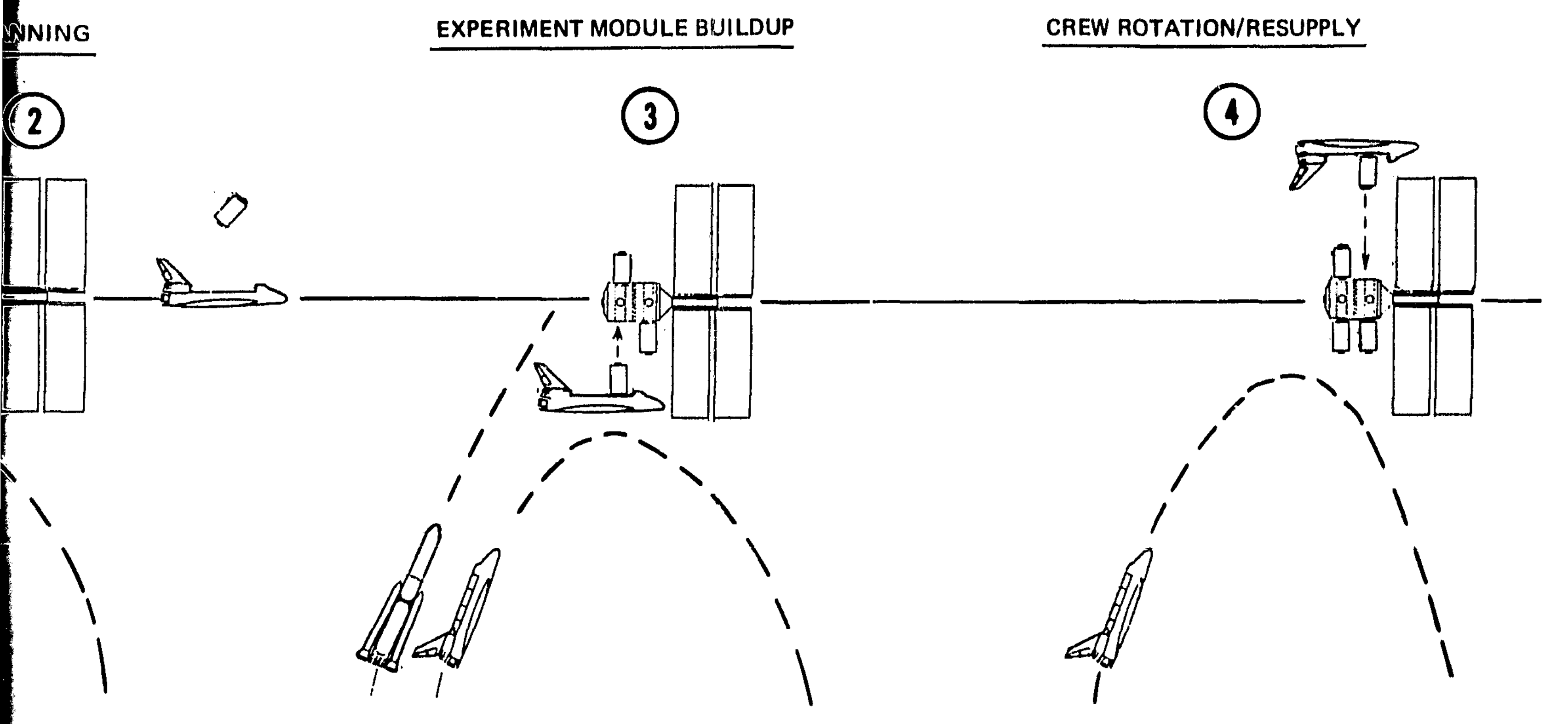
FOLDOUT FRAME



PHASE TIME
CUM TIME

1	2	
• DELIVER STATION WITH HLV	• DELIVER INITIAL 12 MAN CREW	• DE DE • 4 S • 2

FOLDOUT FRAME



2	3	4
IAL 12 MAN	• DELIVER ATTACHED AND DETACHED EXPERIMENT MODULES • 4 SS FLIGHTS PER YEAR • 2 OR 3 HLV FLIGHTS	• DELIVER RESUPPLY MODULE AT 3 MONTH INTERVALS • DELIVER CREW REPLACEMENTS AT 6 MONTH INTERVALS • TOTAL OF 6 SS FLIGHTS PER YEAR

Figure 3.1-5 . Unitary Earth Orbit Space Station Sequence

FOLDOUT FRAME

(unitary-station size) modules launched unmanned by a heavy-lift vehicle. Manned presence for docking and assembly is highly desirable and will require shuttle launches in addition to the station launches. The alternative is to provide automated docking capability for the station modules, including the capability to select by programmed instructions or other means from several available docking ports.

Evacuation of the stations, if required, will normally be accomplished by shuttle rescue. Evacuation of the space base will require several shuttle flights. It is presumed that the space base will include sufficient redundancy and emergency supplies that the time required for evacuation by up to five shuttle flights will be acceptable. Alternatively, a large crew transfer module for rescue (capable of accommodating the entire 60-man crew) could be provided.

Control Functions and Requirements–No requirements outside the normal capabilities of the shuttle or a HLLV were found.

Network Support–Normal shuttle operational procedures are applicable. The stations can maintain continuous communications with the network through TDRSS. Most mission data will be in magnetic tape or hard copy form and will be returned by crew rotation/resupply flights.

3.1.1.4.2.4 Earth Launch Requirements Summary

A total of nine shuttle flights will be required to deliver the modular station modules and crew. One HLLV using 2 SRB's is required to deliver the unitary station (no crew).

Annual modular station transportation needs include resupply flights every three months including a combination resupply/crew rotation flight every 6 months for a total of four shuttle flights. On the combination flight, an OMS kit will be required to satisfy the payload requirement. The unitary station requires resupply flights every two months. A combination crew rotation/resupply occurs at six month intervals for a total of six shuttle flights. An average of four shuttle flights per year will also be required to deliver the application and science modules for both station concepts.

3.1.1.4.2.5 Other Factors

Impact of an operational space manufacturing system on the space stations was briefly investigated. It appears that the scale of operations initially anticipated could be accommodated by a single ASM and therefore would have comparatively little direct impact on station size or configuration. Some of the space manufacturing options require significant power levels (25 to 50 kw), roughly doubling the power output requirements for the station. Placement of a station in a fully illuminated sun-synchronous orbit to provide continuous solar-panel power has been suggested. This orbit,

however, may be unsuited to alternative station missions and reduces shuttle performance capability. The advantage of the fully illuminated orbit must be traded with the increased power-system size and increased launch capability associated with lower inclination orbits.

Space manufacturing, when the technology is mature, is expected to be a highly elastic market. Round trip transportation-cost reductions that may develop through shuttle growth or other means could perhaps lead to a market much larger than projected; a large dedicated station would therefore be required. The likelihood of such a development cannot be predicted with any confidence at this time, but it is recognized as a possible eventuality that would create a significant demand for payload return to Earth.

3.1.1.4.3 Transportation Options Comparison and Evaluation

Not applicable.

3.1.2 60-MAN SPACE BASE

3.1.2.1 Mission Summary

3.1.2.1.1 General Description

The purpose of the 60-man space base is to conduct a more extensive multidisciplinary research, development, and operations program than that provided by the 12-man space station. A total of 38 crew men are dedicated to the space operations and scientific investigations (SOSI) program (called research and application program in 12-man space station concepts). The SOSI provisions include a great variety of internal equipment as well as attached and detached modules.

3.1.2.1.2 Mission Assumptions and Constraints

Mission assumptions and constraints in the reference were the same as for the 12-man modular and unitary stations except for crew size. Recent studies have indicated that an additional potential mission for a space-base sized station is to support crews involved in orbital assembly of very large space systems; see paragraph 3.10.

3.1.2.2 Mission Systems Description

3.1.2.2.1 Mission Options

A single option is described the 60-man space base.

The flight configuration for the space base is shown in schematic form in figure 3.1-6. The configuration consists of six large manned modules; two nuclear power reactors; and five

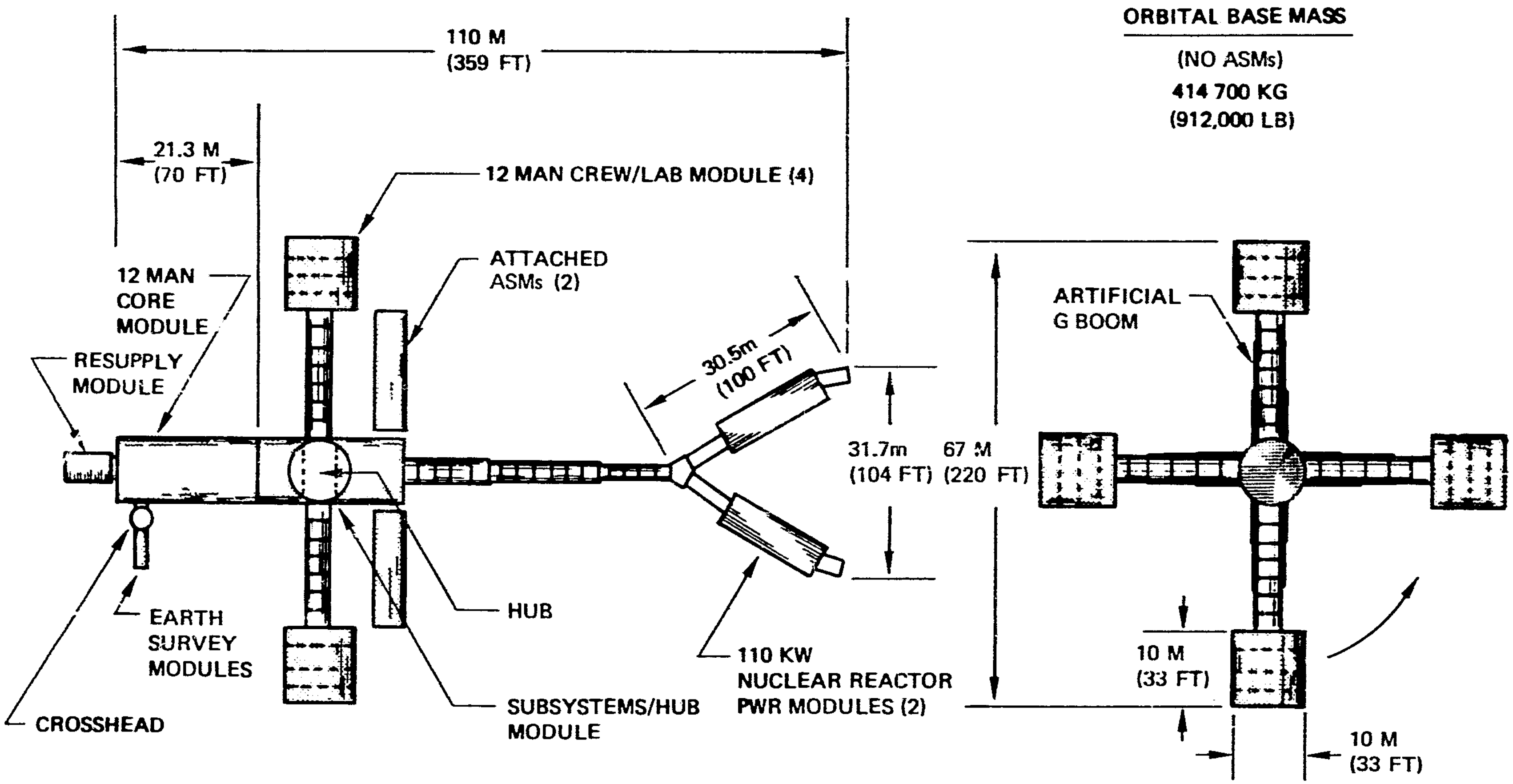


Figure 3.1-6. 60-Man Space Base

telescoping, connecting booms. The space base is defined in the reference was designed to provide both zero g and artificial g capability simultaneously.

Zero g capability is provided by a nonrotating center core; artificial g is provided through use of a hub rotating the four outer modules. Should the space base be operated strictly in a zero g mode, the only major difference would be the reduction in length or possible elimination of the artificial g booms.

The total mass of the basic space base excluding attached and detached modules is 414 700 kg (912,000 lb) as defined in the reference study.

3.1.2.2.2 Payload Descriptions

3.1.2.2.2.1 Base Module Payloads

A summary of the characteristics associated with the major elements of the basic space base is presented in table 3.1-8.

The core module provides crew quarters for 12, a 930-m^2 (10,000-ft^2) solar-array system for use until the nuclear-reactor power systems are in orbit, and a SOSI lab. The subsystem module contains many of the basic station subsystems, another SOSI lab, and the rotating hub for the artificial g modules. Each of the 4 artificial g modules can accommodate 12 crewmen and also include SOSI facilities. Each of the manned modules is similar to the core module of unitary space station in that each is divided into two separate pressure compartments, each provides crew quarters and subsystem for 12 men, and each is 10.06m (33 ft) in diameter. The core and subsystem modules for the space base have 8 decks rather than four as in the unitary station; the artificial g modules include four decks.

The major design difference in these modules as compared to the unitary module is the use of flat-end bulkheads of toroidal bulkheads.

Two zirconium hydride nuclear-reactor modules, each rated at 110 kw, operate normally at one half capacity. A Brayton cycle is used for the conversion system. Four telescoping booms are required for the artificial g modules, and one boom connects the base and the nuclear reactors. A crosshead (small cylindrical structure) module provides docking capability for the Earth survey modules.

3.1.2.2.2.2 Space Operations and Scientific Investigation Payloads

Approximately 31 750 kg (70,000 lb) of SOSI equipment and facilities occupy 10.5 decks within

Table 3.1-8. *Space Base Elements*

ITEM	DESCRIPTION	QTY	UNIT MASS		UNIT SIZE (D x L)	
			10^3 KG	10^3 LB	M	FT
1. CORE MODULE	CREW QUARTERS FOR 12 SOLAR ARRAY SOSI FACILITY	1	76.8	169	10. x 21.3	33 x 70
2. SUBSYSTEM MODULE	SUBSYSTEMS HUB SOSI FACILITY	1	76.4	168	10.1 x 21.3	33 x 70
3. NUCLEAR POWER SYSTEM	ZIRCONIUM HYDRIDE REACTOR BRAYTON CONVERSION SYS	2	36.4	80	4.9 x 30.5	16 x 100
4. ARTIFICIAL G MODULES	CREW QUARTERS FOR 12 SOSI FACILITY	4	40	88	10.1 x 10.7	33 x 35
5. ARTIFICIAL G BOOMS	CONNECT ZERO G CORE WITH ARTIFICIAL G MODULES TELESCOPING	4	4.1	9	4.3 x N/D	10 x N/D
6. NUCLEAR POWER BOOM	CONNECT REACTORS WITH ZERO G CORE	1	6.8	15	4.3 x N/D	10 x N/D
7. CROSSHEAD MODULE	ATTACHMENT FACILITY FOR EARTH SURVEY MODULES	1	5.5	12	4.3 x N/D	10 x N/D
	TOTALS		414.7	912		

the manned modules. The disciplines and technologies provided by these facilities in the reference include the following:

Biomedicine	Materials Processing
Bioscience	Physics
Chemistry	Earth Surveys
Astronomy	General Support

The characteristics of the attached and detached SOSI modules are much the san as those defined for the modular unitary stations, although more modules could be used because of the larger crew size. In general, these modules cover the disciplines of astronomy, Earth surveys, chemistry, and physics.

3.1.2.2.2.3 Crew Transfer and Resupply Payloads

The crew transfer and resupply concepts for the space base are the same as for the modular 12-man space station. Crewmen stay a maximum of six months and are normally exchanged 12 men at a time.

Resupply requirements per month are considerably greater than for the 12-man stations and amount of approximately 15 150 kg (33,400 lb) delivered to the base and 6 580 kg (14,500 lb) returned. Included within this total are 8 570 kg (18,900 lb) for crew consumables and station expendables. The remainder of the resupply mass is related to experiment equipment. None of the resupply masses identified above include mass for the RM. Using the same size resupply module as in the 12-man station concepts, a resupply mission would be required every 2 weeks. If the RM were enlarged to take advantage of all of the shuttle capability, the resupply interval would be 2-1/2 to 3 weeks depending on the orbit.

3.1.2.2.2.5 Mass Summary

The mass summary of the space base is summarized in table 3.1-9. Resupply payloads are similar to those described for the 12-man stations (paragraph 3.1.1.2.2.3 et seq.).

Table 3.1-9. Space Base Mass Summary*

Payload	Mass	
	10^3KG	10^3LB
Basic station	414.7	912
Attached ASM	25.4	60
Resupply module	10	22
Crew	5.5	12
Total	455.5	1005.9

* Without growth

Resupply payloads are similar to those described for the 12-man stations (paragraph 3.1.1.2.2.3 et seq).

3.1.2.2.2.6 Pickup Points and Transportation Constraints

Base Modules—These are the same as for the 12-man unitary station (paragraph 3.1.1.2.2.6).

ASM's, SOSI Modules, CTV's, RM's—These are handled as for the 12-man stations (paragraph 3.1.1.2.2.6).

3.1.2.2.3 Transfer and Storage

Resupply modules (RM's) provide necessary storage for consumables—transfer is not ordinarily required. Some ASM's or SOSI modules are likely to require transfer of fluids from the RM's. If the space base is used to support assembly operations in orbit, it must provide docking and storage for payloads awaiting use, and for orbital assembly equipment and tools. This requirement may be expected to change the station configuration from that described.

3.1.2.2.4 Orbital Assembly, Maintenance, and Modification

The space base is to be assembled in orbit from large modules by docking them together in the prescribed configuration. The modules cannot be returned to Earth; therefore maintenance and modification will be carried out by the base crew in orbit. A possible use of the base to support major orbital assembly operations is described in paragraph 3.10

3.1.2.3 Transportation Requirements

3.1.2.3.1 Payload Delivery Points

All payloads for the space base are delivered to low Earth orbits. Orbit-to-orbit stages are not required. Orbit inclinations from 28-1/2° to sun synchronous (about 97°) have been examined for space stations. No selection has been made. It is quite likely that stations intended for different purposes will be flown at different inclinations.

3.1.2.3.2 Payload Delivery Options

Base modules are sized for a HLLV transportation system. The referenced study assumed use of a two-stage Saturn V and hence configured 10m (33 ft) diameter modules. Shuttle derived HLLV's are typically 8.2m (27 ft) in diameter and the base modules would be redefined to that size.

Delivery of the space base payloads is assumed to use space shuttle and a heavy-lift launch vehicle. Payload mass growth allowances for transportation analysis employed 24% on the CTM, RM, and

ASM's; they are similar to those of the 12-man stations and have been studied in considerable detail. A 37% factor was applied to the basic space base elements including the manned modules, nuclear reactors, and connecting booms.

3.1.2.3.2.1 Base Module and Equipment Delivery

The reference study concept for delivery of the manned modules was to launch eight decks either a single 8-deck module or two 4-deck modules. With this approach and the designated growth factor applied the delivery mass for each launch would be approximately 97 500 kg (215,000 lb). An option to this approach would be the delivery of just four-deck modules, reducing the delivery mass to an average of approximately 45 360 kg (100,000 lb).

Artificial g booms, the nuclear power boom, and the crosshead have delivery masses ranging between 5 440 kg (12,000 lb) and 9 530 kg (21,000 lb).

3.1.2.3.2.2 Crew Transfer/Resupply Delivery

Modules for these functions will be delivered by the space shuttle and are the same as for the unitary space station.

3.1.2.3.2.3 SOSI Module Delivery

The attached modules will have a nominal delivery mass of 8 440 kg (18,600 lb) and nominal size 4.3 x 8.23m (14 x 27 ft). The maximum attached module would be 45 800 kg (101,000 lb) and 7.6m x 20.7m (25 ft x 68 ft). The detached SOSI modules have a mass range of 12 250 kg (27,000 lb) to 29 480 kg (65,000 lb) and a size range of 20.4m to 25.3m (67 ft to 83 ft). All had diameters of 4.3m (14 ft).

3.1.2.3.2.3 Operational Constraints

Normal shuttle operational procedures are applicable. HLLV launches will be unmanned.

3.1.2.4 Mission/Transportation Modes and Operations

3.1.2.4.1 Transportation Options

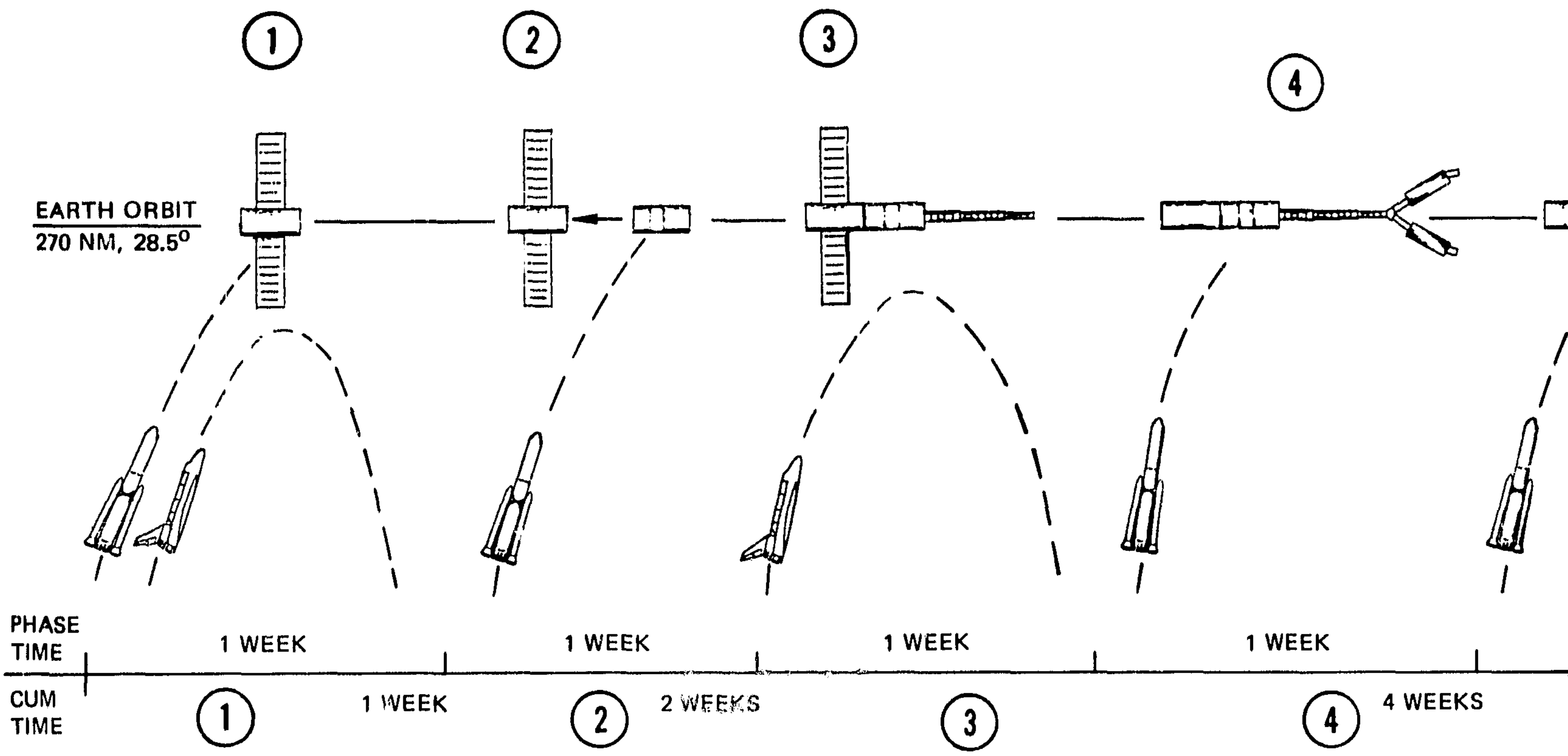
A heavy lift launch vehicle (HLLV) employing modified components of the space shuttle and the space shuttle itself are required to perform this mission.

3.1.2.4.2 Representative Transportation Mode and System

3.1.2.4.2.1 Transportation Sequence

The major transportation operations involved in the space base build-up and its subsequent

STATION BUILD-UP



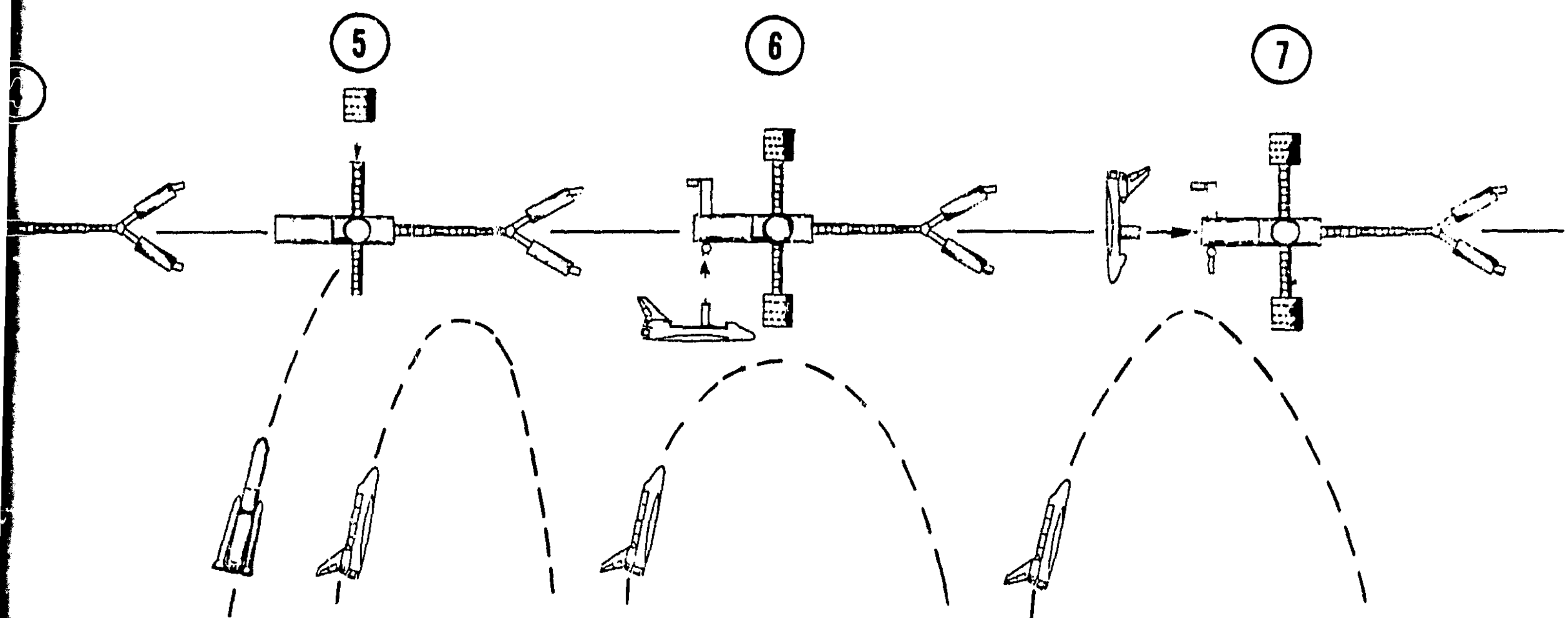
• DELIVER CORE MODULE WITH HLV • DELIVER INITIAL 12 MEN WITH SS • DEPLOY SOLAR ARRAY AND ACTIVATE SUBSYSTEMS	• DELIVER SUBSYSTEM/ HUB MODULE WITH HLV • DOCK WITH CORE MODULE • CHECKOUT ASSEMBLED CONFIGURATION	• DELIVER TELESCOPING BOOM FOR NUCLEAR POWER MODULES WITH SS • DOCK TO SUBSYSTEM/HUB MODULE	• DELIVER NUCLEAR POWER MODULES (2) WITH HLV • STOW SOLAR ARRAY

FOLDOUT FRAME

1

ON BUILD-UP

CREW ROTATION/RESUPPLY



EEK | 4 WEEKS

4 WEEKS | 8 WEEKS

4	5	6	7
ER NUCLEAR MODULES (2) HLV SOLAR ARRAY	• DELIVER TELESCOPING BOOMS FOR CREW/LAB MODULES WITH SS (2 LAUNCHES) • DELIVER 4 CREW/LAB MODULES WITH HLV (2 LAUNCHES) • DELIVER 24 CREWMEN WITH SS AFTER EACH PLACEMENT OF 2 CREW/LAB MODULES (2 LAUNCHES)	• DELIVER EXPERIMENT MODULE BOOM • DELIVER ATTACHED AND DETACHED EXPERIMENT MODULES • 8 SS FLIGHTS PER YEAR • 2 OR 3 HLV FLIGHTS	• DELIVER RESUPPLY MODULE AT 1 MONTH INTERVALS WITH SS • DELIVER CTM (12 CREWMEN) AT 5 WEEK INTERVALS WITH SS

FOLDOUT FRAME 2

Figure 3.1-7. Space Base Mission Transportation Sequence

operations are shown in figure 3.1-7. The HLLV is used to deliver the large base modules while the space shuttle delivers crewmen, supplies, ASM's and connecting booms.

An alternate transportation mode for resupply would be to have the supplies delivered by a HLLV at 6 month intervals thereby eliminating the 12 shuttle resupply flights.

3.1.2.4.2.2 Transportation Sizing

Not applicable.

3.1.2.4.2.3 Operational Factors

Sequence operations are the same as for the 12-man stations (paragraph 3.1.1.4.2.4) except that more frequent launches occur.

3.1.2.4.2.4 Earth Launch Requirements Summary

Five HLLV's (4 SRB) are required to deliver the major elements of space base. One space shuttle flight is required to transport the nuclear power boom. Annual space shuttle requirements would include 12 for resupply, 10 for crew rotation (one every 5 weeks), and at least 4 for ASM delivery.

The resupply requirement could also be satisfied by using two HLLV (2 SRB) launches instead of 12 shuttle flights.

3.1.2.4.2.5 Other Factors

If the space base is used to support assembly of large space systems such as power satellites, the launch requirements of delivery of those payloads will be much greater than those stated here. See paragraph 3.10 for typical data.

3.1.2.4.3 Transportation Options Comparison and Evaluation

Not applicable.

3.2 GEOSYNCHRONOUS OPERATIONS PROGRAM

Geosynchronous orbit is the preferred location for a number of science and applications space operations. Examples of these operations include weather reconnaissance, communications/ navigation, global environmental science, Earth survey, and the generation or transmission of electrical power for terrestrial needs by satellite energy systems. Conduct of these operations may be accomplished through use of manned space stations and automated spacecraft.

The missions selected to represent advanced geosynchronous transportation requirements are a manned space station and a manned sortie to service automated spacecraft.

Two other missions were briefly investigated: a large direct-access (mass) communication satellite, and a large space telescope (LST) with an auxiliary subsatellite for attempts at observation of planets around nearby stars. The LST payload would most probably be placed at a lunar libration point rather than geosynchronous orbit but is included here because the transportation requirements are similar to those for geosynchronous transfer. These two missions are discussed in paragraph 3.2.3.

Analysis of satellite energy systems are discussed separately in Section 3.10 due to the uniqueness of that program option.

3.2.1 GEOSYNCHRONOUS SPACE STATION

3.2.1.1 Mission Summary

3.2.1.1.1 General Description

The reference geosynchronous space station (GSS) mission consists of a modular station that can be continuously occupied by a crew of eight and can accommodate both Earth application and science sensors.

The concept is illustrated in figure 3.2-1. System elements making up this concept include the basic modules, application and science modules, crew transfer vehicle and resupply modules. A unitary (single module) station option is also described.

The station provides quarters for the eight-man crew, supporting subsystems and consumables. The functions provided are as follows: housing basic station subsystems; crew quarters for eight men and sixteen in an emergency; command/control centers and radiation shelter; electrical power; galley and recreation; and cryogenics and other storage.

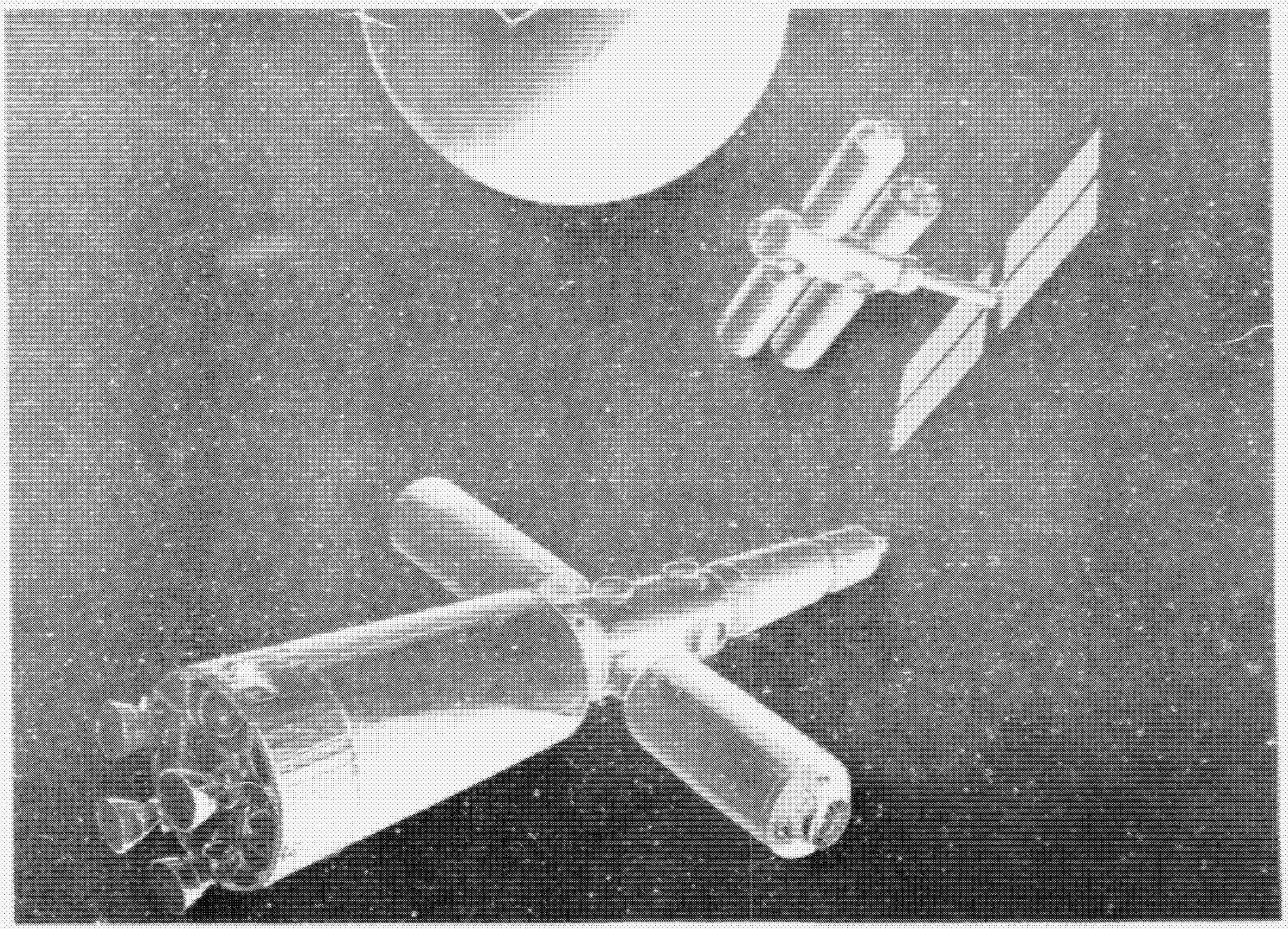


Figure 3.2-1. Geosynchronous Space Station Delivery

3.2.1.1.2 Mission Assumptions and Constraints

Table 3.2-1 summarizes mission assumptions and constraints. The geosynchronous orbit location assumption is not applied to the LST option payload as noted above.

Table 3.2-1: Geosynchronous Mission Assumptions and Constraints

MISSION	OBJECTIVES	MISSION ASSUMPTIONS & CONSTRAINTS
GEOSYNCHRONOUS OPERATIONS, INCLUDING MANNED	● SUPPORT VARIOUS OPERATIONS IN GEOSYNCHRONOUS EVOLUTIONARY PROGRAM 1. WEATHER RECONNAISSANCE 2. AIRCRAFT TRACK 3. GLOBAL/ENVIRONMENTAL SCIENCE 4. COMMUNICATIONS ● TELEPHONE/TV (CONVENTIONAL) ● MASS COMMUNICATIONS 5. SERVICING AUTOMATED GEOSYNCHRONOUS SPACECRAFT 6. VARIOUS SCIENCE MISSIONS & PAYLOADS	● INITIAL AUTOMATIC PLACEMENT AND OPERATION ● GEOSYNCHRONOUS ORBIT LOCATION ● CAPACITY FOR UP TO EIGHT PERSONS ● CREW EVA CAPABILITY ● DOCKING SYSTEM FOR PLUG-IN MODULES, INCLUDING LOGISTICS (RESUPPLY) MODULES ● SIZE/FUNCTION ADAPTATION BY MODULAR ADDITIONS

3.2.1.2 Mission Systems Description

3.2.1.2.1.1 Modular Geosynchronous Space Station (MGSS)

The flight configuration for the MGSS is shown in figure 3.2-2. Nine station modules are required to provide quarters for the eight-man crew, supporting subsystems, and consumables. Three application and science modules (ASM's) are permanently attached to the station; at least one detached ASM flies in close formation and is refurbished at the station. The two crew transfer vehicles (CTV's) provide immediate emergency survival capability to allow time for a rescue mission. They are also used for crew rotation.

Crew rotation occurs every 6 months by bringing up the crew in a new CTV with the replaced crewmen returning in a CTV already at the station. Station resupply is also nominally performed every 6 months with the use of a resupply module (RM). Following integration of the new RM into the station system, the expended RM is returned for refurbishment and subsequent reuse on another resupply mission.

The total mass of the MGSS with three ASM's and two CTV's is approximately 125 000 kg (275,000 lb).

3.2.1.2.1.2 Unitary Geosynchronous Space Station (UGSS)

The flight configuration for the unitary GSS is shown in figure 3.2-3. The major difference between this concept and the modular GSS is that a single module provides the required volume for the crew of eight, the subsystems, and the consumables. All other features—including the application and

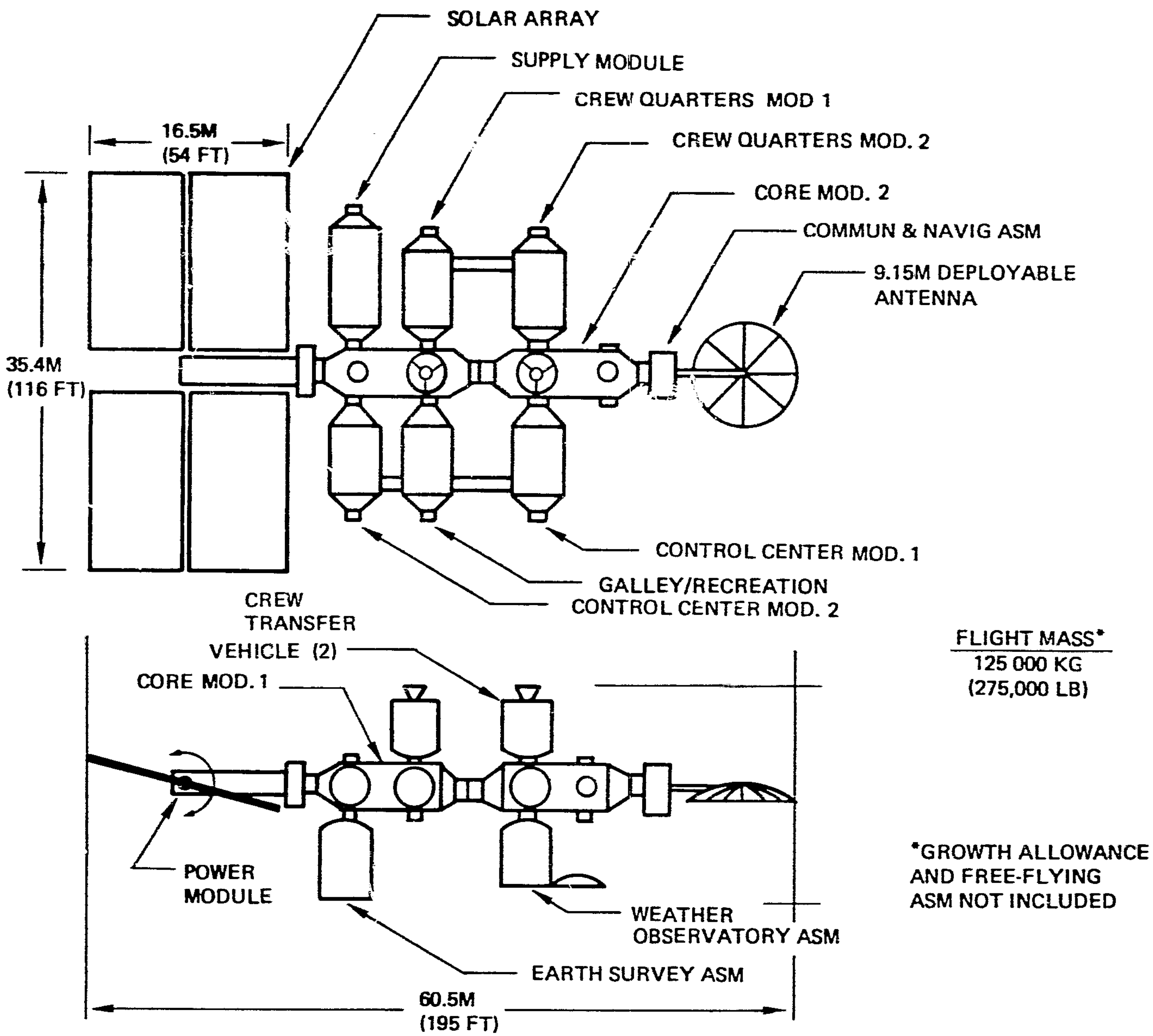


Figure 3.2-2. GSS Modular Flight Configuration

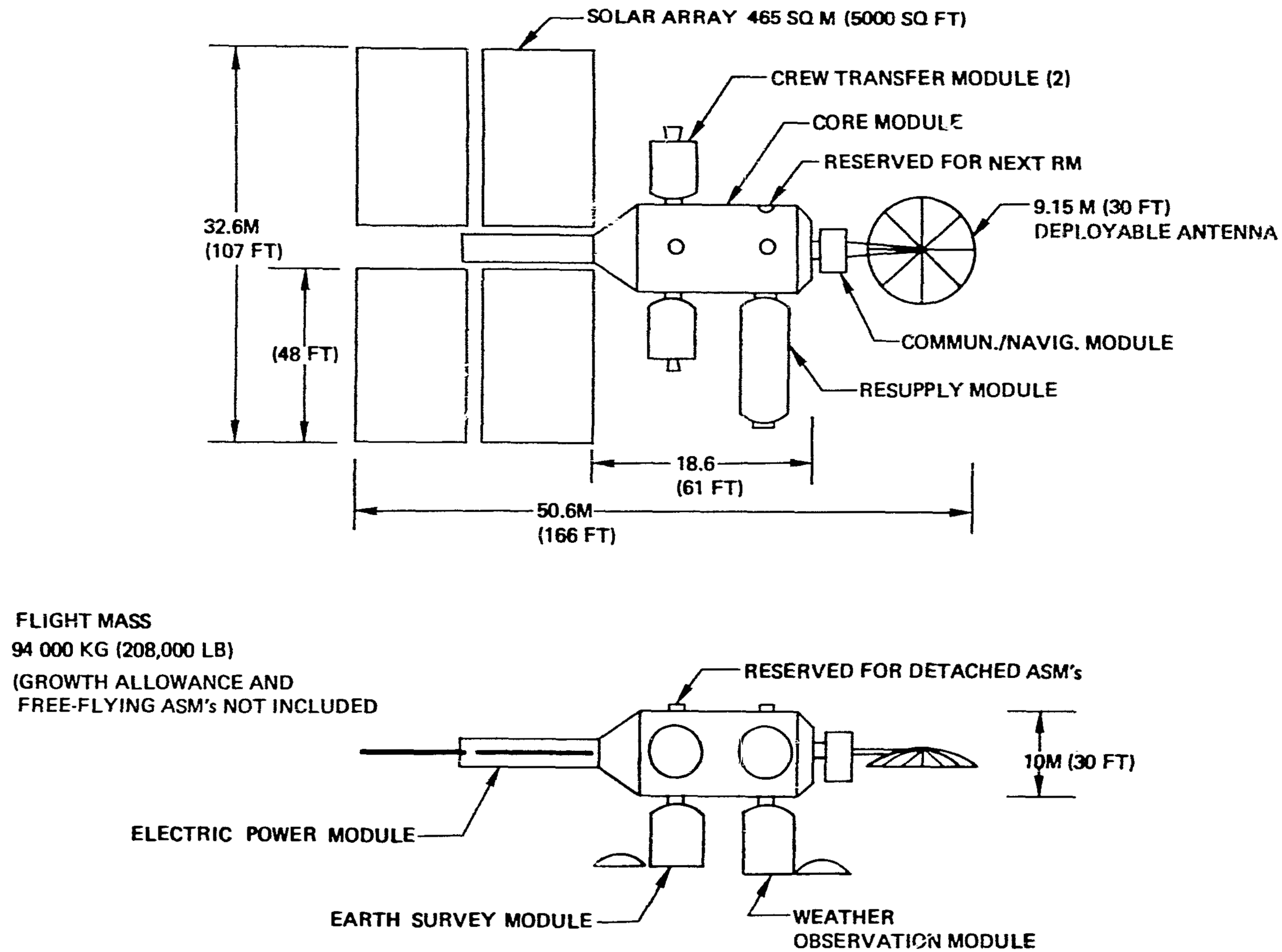


Figure 3.2-3. Unitary Geosynchronous Space Station

science modules, crew transfer vehicles for emergency abort and crew rotation, and resupply provisions are the same. The total orbital mass of the unitary GSS with two CTV's and three ASM's is approximately 95 000 kg (209,000) lb) as identified by the reference.

3.2.1.2.2 Payload Descriptions

3.2.1.2.2.1 Modular Geosynchronous Station

A brief description of the size and masses of the nine modules forming the station is presented in table 3.2-2. Since both are designed to support eight crewmen from the beginning, the modules selected use the same functional design approach as the modular OLS. The center of gravity location for each module is estimated to be at its geometric centroid.

Table 3.2-2. Modular Station Payloads

ITEM	DESCRIPTION	UNIT MASS (INCLUDING CONSUMABLES)		UNIT SIZE (DXL)	
		10^3 KG	10^3 LB	M	FT
1. CORE MODULE-1 (CM-1)	CENTRAL KEEL BASIC STATION SUBSYS. IVA/EVA AIRLOCK	14.5	32	4.3 x 12.8	14 x 42
2. CORE MODULE-2 (CM-2)	GENERALLY SAME AS CM-1	11.3	25	4.3 x 12.8	14 x 42
3. ELECTRICAL POWER MODULE (EPM)	SOLAR ARRAY-465M^2 CRYO STORAGE	9.5	21	4.3 x 10	14 x 32
4. SUPPLY MODULE (SM)	CRYO STORAGE CRYO CARGO STORAGE	16.5	39	4.3 x 11.3	14 x 37
5. CONTROL CENTER MODULE-1 (CCM-1)	BACK-UP CONT. CENTER RADIATION SHELTER LABS	16.8	37	4.3 x 9.8	14 x 32
6. CONTROL CENTER MODULE-2 (CCM-2)	PRIMARY CONT. CENTER EXERCISE/MEDICAL	5.4	12	4.3 x 9.8	14 x 32
7. CREW QUARTERS MODULE-1 (CQM-1)	4 STATEROOMS ECLSS FOR 8 MEN BACK-UP GALLEY	6.8	15	4.3 x 9.8	14 x 32
8. CREW QUARTERS MODULE-2 (CQM-2)	GENERALLY SAME AS CQM-1 INCLUDES COMMANDERS S.R.	5.9	13	4.3 x 9.8	14 x 32
9. GALLEY/RECREATION MODULE (GRM)	GALLEY & DINING RECREATION	7.3	16	4.3 x 9.8	14 x 32
	TOTALS	94	207		

Two crew quarters modules (CQM 1 and 2) are included with each providing four staterooms and sufficient ECLS equipment to accommodate eight crewmen. These modules also have dual functions. The CQM 1 provides a backup galley and CQM 2 includes the station commander's stateroom. The final module provides the galley and dining facilities as well as recreation.

The two core modules (CM 1 and 2) house basic station subsystems (such as G&N and RCS) and provide the framework to allow attachment of all other modules. The electrical power module (EPM) supports the 465-m^2 (5,000-ft^2) solar array that provides the average load of approximately 20 kw. Also included is cryogenic storage for fuel cells and batteries. Although the GSS has approximately the same power requirements as the OLS, the solar array size is only about one-half since the occultation periods are approximately 5% rather than 38%. The supply module (SM) contains the majority of the cryogenics and bulk cargo for the station. It houses 120 days of supplies.

Control of the GSS is provided from either of the control center modules (CCM 1 and 2); CCM 1 also provides the radiation shelter and science labs; and CCM 2 also serves as the medical and exercise facility.

3.2.1.2.2.2 **Unitary Station**

The basic unitary station for geosynchronous operation consists of a core module and power module. The combined length of the two modules in the launch configuration is 30.8m (101 ft) and the mass (including consumables, and inert fluids) is 70 200 kg (154,800 lb). Physical data are summarized in table 3.2-3. The center of gravity of the station is estimated to be roughly 2m (6.6 ft) forward, i.e., toward the power module, of the geometric centroid of the core module.

Table 3.2-3. Unitary GSS Payloads

ITEM	DESCRIPTION	UNIT MASS INCLUDING CONSUMABLES		UNIT SIZE (D x L)	
		10^3 KG	10^3 LB	M	FT
CORE MODULE	ALL FUNCTIONS EXCEPT ELECTRIC POWER	60.7	133.8	8.23 x 18.6	27 x 61
POWER MODULE	SOLAR/ELECTRIC POWER SUPPLY	9.54	21	4.3 x 10	14 x 32
	TOTALS	70.2	154.8		

The core module is essentially the same as that used for the unitary orbiting lunar station. The major differences between this core module and that used in the unitary low-Earth-orbit space station are a diameter of 8.23m (27 ft) versus 10m (33 ft) and the inclusion of a radiation shelter in the GSS. The length of the core module is 18.6m (61 ft). The dry mass is 38 600 kg (85,100 lb); the consumables and inert fluids amount to 22 100 kg (48,700 lb).

The power module for the UGSS includes a 465-m^2 (5,000-ft^2) solar array that is the same as for the modular GSS. This module can be separated from the core module if required. The length of the module is 10m (32 ft) and the total mass is 9 540 kg (21,000 lb).

3.2.1.2.2.3 Applications and Science Modules

Four ASM's were selected by the referenced study for use at the GSS. This number is representative of the transportation requirement and is used in this study. The ASM's are used with either station option. The characteristics of these modules are summarized in table 3.2-4.

Table 3.2-4. GSS Application and Science Payloads

ITEM	DESCRIPTION	QTY	UNIT WEIGHT		UNIT SIZE (DXL)	
			KG	LBS	M	FT
1. WEATHER OBSERVATORY MODULE (WOM)	ATTACHED. IMMEDIATE & CONTINUOUS	1	5,455	12,000	4.3 x 7.9	14x26
2. EARTH SURVEY MODULE (ESM)	ATTACHED, GEOGRAPHIC & TOPOGRAPHIC	1	7,270	16,000	4.3 x 7.9	14x26
3. COMMUNICATION/NAVIG. MODULE (CNM)	ATTACHED. TV TRANSMISSION OF WEATHER. 9.15 DIA (30 FT) ANTENNA	1	2,270	5,000	4.3 x 7.9	14x26
4. SOLAR ASTRONOMY MODULE (SAM)	DETACHED, EXCEPT FOR REFURBISHMENT EVERY 90 DAYS	1	7,770	17,100	4.3x17.7	14x58
	TOTAL		22 765	50,100		

Three of the four ASM's specified are permanently attached to the station. The weather observatory module (WOM) provides the required sensors and data analysis capability to allow continuous weather coverage including real-time severe-storm warning. TV transmission of much of the weather data is accomplished by the communication and navigation module (CNM) using a large deployed antenna. Immediate coverage of geographic features (such as population centers, topographic regions, natural resources, etc.) is possible through use of optics and other sensors in the Earth survey module (ESM).

The detached ASM depicted is a solar astronomy module (SAM). Deployment at the geosynchronous altitude will provide continuous viewing of the sun rather than the intermittent viewing from deployment in low Earth orbits. It is also conceivable that more than one type of SAM would be desirable. Refurbishment of the detached modules will take place at the station approximately every 90 days.

Center of gravity for the ASM's is estimated to be at the geometric centroid of each.

3.2.1.2.2.4 Crew Transfer and Resupply Payloads

Crew rotation is expected to occur at least every 6 months in order to not exceed the allowable radiation dose. Normally the entire crew is changed at each rotation period. Resupply of crew consumables and station expendables is also scheduled at 6-month intervals. A summary of the characteristics of the modules used to perform the crew transfer and resupply functions is presented in table 3.2-5.

Table 3.2-5. Crew Transit and Resupply Payloads

ITEM	DESCRIPTION	UNIT WEIGHT		UNIT SIZE (DXL)	
		10^3 KG	10^3 LB	M	FT
CREW TRANSFER VEHICLE (CTV)	NOMINALLY- 8 MEN EMERGENCY-8 MEN 30 DAYS SUPPLIES/SUBSYS	8.0 UP AND DOWN	17.5	4.3 x 4.3	14 x 14
RESUPPLY MODULE	CREW CONSUMABLES STATION EXPENDABLES	15.2 UP 5.2 DOWN	33.4 UP 11.4 DOWN	4.3 x 11.3	14 x 37

Crew Transfer Vehicle (CTV)—The primary function of the CTV is to house the crew and supporting subsystems during the transit from low Earth orbit to synchronous orbit and return back to low Earth orbit. Transportation is normally provided by an orbit transfer vehicle (OTV).

One section of the CTV provides shirt-sleeve environment for the crew and also houses a portion of the required subsystems. This section is sized to normally accommodate four crewmen although eight is possible in an emergency (such as the second CTV at the station not operating properly). An unpressurized section of the CTV contains the remainder of the subsystems. In the nominal abort

mode, the CTV is used to provide emergency shelter for the crew until a rescue can be accomplished. It therefore is sized for 30 days' emergency supplies. This requires less mass than provision of a propulsion system capable of a total delta V of 4 200 m/sec (14,000 ft/sec) for return back to a low Earth orbit.

Resupply Module—The resupply module (RM) is essentially the same as the supply module initially launched with the station. These modules are sized to accommodate the supplies required for the nominal 6-month resupply interval plus 30 days contingency. Transportation between orbits is provided by an OTV. The unitary station carries initial consumables internally.

A pressurized section contains 43 m^3 (1,500 ft^3) of volume to store bulk cargo and the unpressurized section is used to house the LO_2, LN_2, LH_2, and N_2H_4 tanks.

When the consumables in the RM are used up, the RM can be loaded with items to be returned to Earth; upon return it is refurbished for reuse at a later time.

The consumables and expendables included in the RM for the 6-month-nominal-plus-30-day contingency total 12 170 kg (26,800 lb). The RM empty mass is estimated as 4 300 kg.

Consumables—Sufficient consumables for 180 days plus 30 days contingency are included in the initial station delivery. Each resupply mission provides capability for the same quantity. The 30 days' contingency supply is replaced only if actually used. Consumables are summarized in table 3.2-6. Values apply to either station option.

Table 3.2-6. GSS Consumables and Expendables 180 Day Nominal + 30 Day Contingency

ITEM	KG	LB
FOOD	1960	4320
ECLSS	326	718
STATION SPARES	2100	4610
STATION CRYOGENCS		
LH_2	680	1490
LO_2	2520	5550
LN_2	2460	5400
EXPERIMENT FLUIDS		
LN_2	74	163
N_2H_4	1840	4050
EXPERIMENT SPARES	210	463
TOTAL	12170	26850

3.2.1.2.2.5 Mass Summary

Table 3.2-7 and Table 3.2-8 present a mass summary for the geosynchronous manned station options. Values do not include weights growth.

Table 3.2-7. Mass Summary for GSS Mission Start-Up

PAYLOADS	MODULAR STATION OPTION		UNITARY STATION OPTION	
	10^3 KG	10^3 LB	10^3 KG	10^3 LB
BASIC STATION	95	210	70.5	155.8
ATTACHED ASM'S	15	33	15.0	33.0
CTV (2)	15.9	35	15.9	35.0
TOTAL STATION ASSEMBLY	125.9	278	101.4	222.8
DETACHED ASM	7.77	17.1	7.77	17.1
TOTAL DELIVERED	133.67	295.1	109.17	239.9

Table 3.2-8. Mass Summary for Crew Rotation/Resupply Mission

PAYLOADS	10^3KG	10^3 LB
CTV (1)	8.0	17.5
RM DRY	2.9	6.4
CONSUMABLES	12.2	26.8
DELIVERY	23.0	50.7
RETURN	13.1	28.9

3.2.1.2.2.6 Pickup Points and Transportation Constraints

Modular Station—These station modules include docking ports at either end that provide pickup points. For transportation in the Shuttle payload bay adapter fixtures will be required to bridge from docking ports to payload bay attach points. Docking ports provide adequate pickup points for orbit-to-orbit transportation. These modules must be protected from aerodynamic loads during Earth launch.

Unitary Station—The large unitary station modules incorporate structural hard points around their aft circumference for adaption to a heavy lift launch vehicle or orbit transfer vehicle payload support structure. Docking ports provide pickup points for module grouping as needed for orbit transfer. The unitary station modules must be protected from aerodynamic loads during Earth launch.

ASM's—These are handled like the modular station modules.

CTV and RM's—These are handled like the modular station modules.

Orbit-to-orbit transfer accelerations must be limited to about 5 m/sec^2 (1/2 g) when modules are grouped in assemblies connected by docking structures.

3.2.1.2.3 Transfer and Storage

Fluids transfers are not required. Fluids are supplied in resupply modules (RM's). These remain connected to the station assemblies until their consumables are exhausted. They can then be returned to Earth for refilling and reuse. Bulk cargoes of small size (food, instruments, etc.) are transferred from RM's to the station manually through docking ports. Large payloads (e.g. ASM's) are provided in modular units with docking capability; transfer as such is not required.

3.2.1.2.4 Orbital Assembly, Maintenance, and Modifications

The stations will initially be assembled in low orbit for checkout. They will then be disassembled as required to suit the selected orbit-to-orbit delivery option [illegible] flight delivery, and reassembled in the mission (geosynchronous) orbit.

All maintenance required will be performed by the crew in the mission orbit. Return of station elements to Earth from geosynchronous orbit for maintenance does not appear practical. Other mission elements can be returned to Earth for maintenance if necessary.

Orbital modification is confined to changeout of experiments and ASM's.

3.2.1.3 Transportation Requirements

3.2.1.3.1 Payload Delivery Points

Transportation requirements include several types of payloads and delivery points. Payload delivery points are low Earth orbit, geosynchronous orbit, and Earth return (either direct or through a low Earth orbit). Requirements are shown in Table 3.2-9

Table 3.2-9. Payload Delivery Points

PAYLOAD	LOW ORBIT	GEOSYNCHRONOUS ORBIT	EARTH RETURN
STATION ELEMENTS	x	x	
ORBIT TRANSFER SYSTEMS	x	x	x
CREW ROTATION AND RESUPPLY	x	x	x
ABORT VEHICLE (IF USED)	x	x	ABORT ONLY
EXPERIMENTS AND FREE FLYER PAYLOADS	x	x	OPTIONAL

3.2.1.3.2 Payload Delivery Options

The modular station is delivered to low Earth orbit one module at a time by the space shuttle; the unitary station is delivered as a single unit by a HLLV. Stations are assembled in low orbit for checkout and then partially disassembled for delivery to the geosynchronous orbit as appropriate to orbit transfer capabilities (see paragraph 3.2.1.3.2.2 below).

A combination crew rotation and resupply mission at 6-month intervals results in the fewest number of flights. Since at least one ASM is delivered with the station, the other three ASM's are included in ASM changeout flights.

Payload values used in sizing the reference and alternate transportation systems include a 24% allowance on hardware mass.

3.2.1.3.2.1 Station Delivery

Station delivery options were developed using several key guidelines: 1) for concepts requiring more than one orbit transfer launch, a crew must be present to assist in the final docking maneuvers; 2) only one-half the crew need be present during the station assembly phase; and, 3) when the crew is present, modules must be present to provide power and control for the station and at least 3 months (plus 1 month reserve) of supplies.

Using these guidelines at least four basic options are possible for the modular GSS.

The two-launch station option shown in figure 3.2-4 was selected as representative. This concept has a manned first launch transporting approximately one-half of the station in its orbital configuration and one-half the crew. The second launch brings up the remainder of the crew and the station (also configuration). Maximum delivery mass is approximately 60 000 kg (133,000 lb).

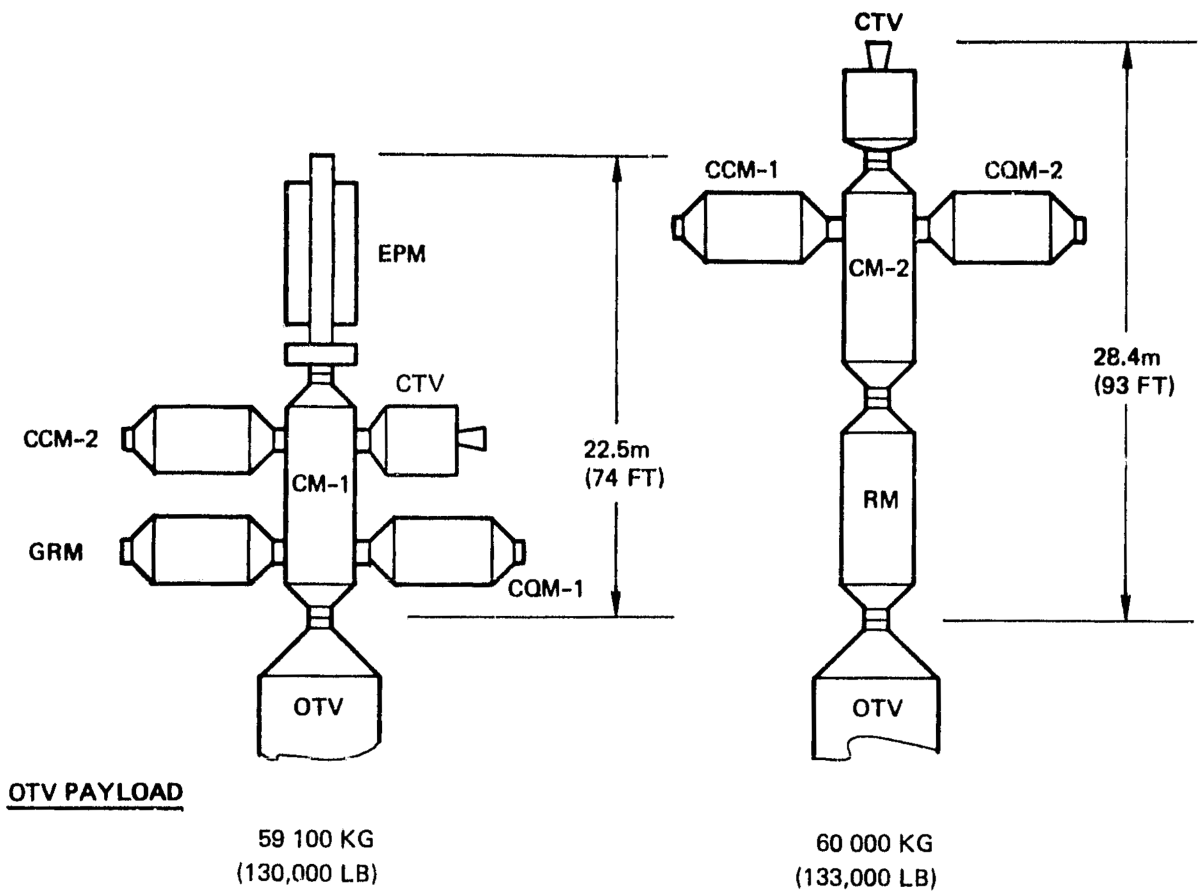


Figure 3.2-4. Modular GSS Delivery—Two Flights

A triple-launch concept can result in payloads of approximately 40 900 kg (90,000 lb) as shown in figure 3.2-5. The first launch is unmanned and provides basic station power, control, and quarters. The second launch is manned and also delivers supplies and the second core module. The third launch is unmanned and brings up the remaining modules.

Use of a single launch to deliver the GSS would result in a payload of approximately 125 000 kg (275,000 lbs) while use of four launches would reduce the payloads to approximately 27 300 kg (60,000 lbs). These options are illustrated in figures 3.2-6 and 3.2-7.

Two delivery options are possible should the unitary GSS be considered, single and two-flight deliveries. The two-flight option is shown in figure 3.2-8. For a one-flight delivery, the OTV payload requirement would be approximately 86 300 kg (190,000 lbs).

3.2.1.3.2.2 Crew Rotation and Resupply

The reference option employs combined crew rotation and resupply. Payloads are shown in figure 3.2-9. A combination crew rotation/ASM/resupply delivery is also

3.2.1.3.2.3 Application and Science Modules

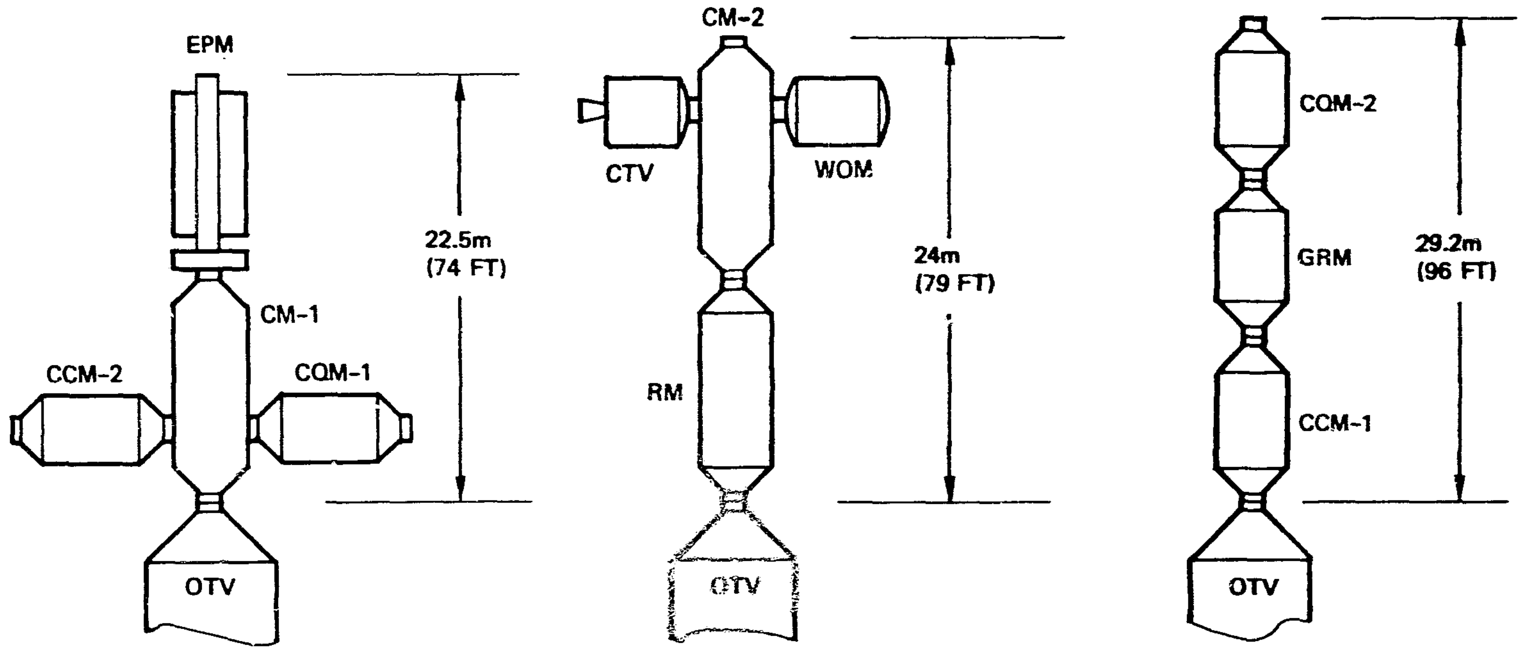
Several options are available for the delivery of ASM's;

1. Launch all four on a separate OTV flight
2. Include one ASM on each crew rotation/resupply flight
3. Launch one with each station cluster (2) with the other two ASM brought up on the first two crew rotation/resupply flights.

The delivery mass of these options is shown in table 3.2-10 Option 1 has been used to define a reference mission.

Table 3.2-10. ASM Delivery for GSS

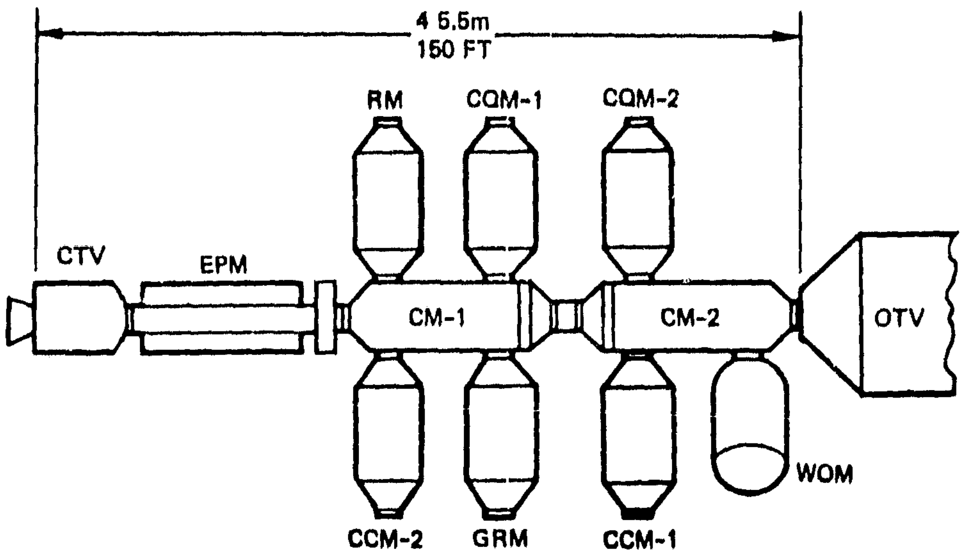
OPTION	DELIVERY		RETURN	
	10^3 KG	10^3 LB	10^3 KG	10^3 LB
OPTION 1	27.3	60	0	0
OPTION 2	31.8	70	15	33
OPTION 3	65.9	145	0	0



OTV PAYLOAD

40,910 KG (90,000 LBS)	40,910 KG (90,000 LBS)	39,000 KG (86,000 LBS)

Figure 3.2-5. Modular GSS Delivery—Three Flights



OTV PAYLOAD 125,000KG (275,000)

Figure 3.2-6. Modular GSS Delivery—One Flight

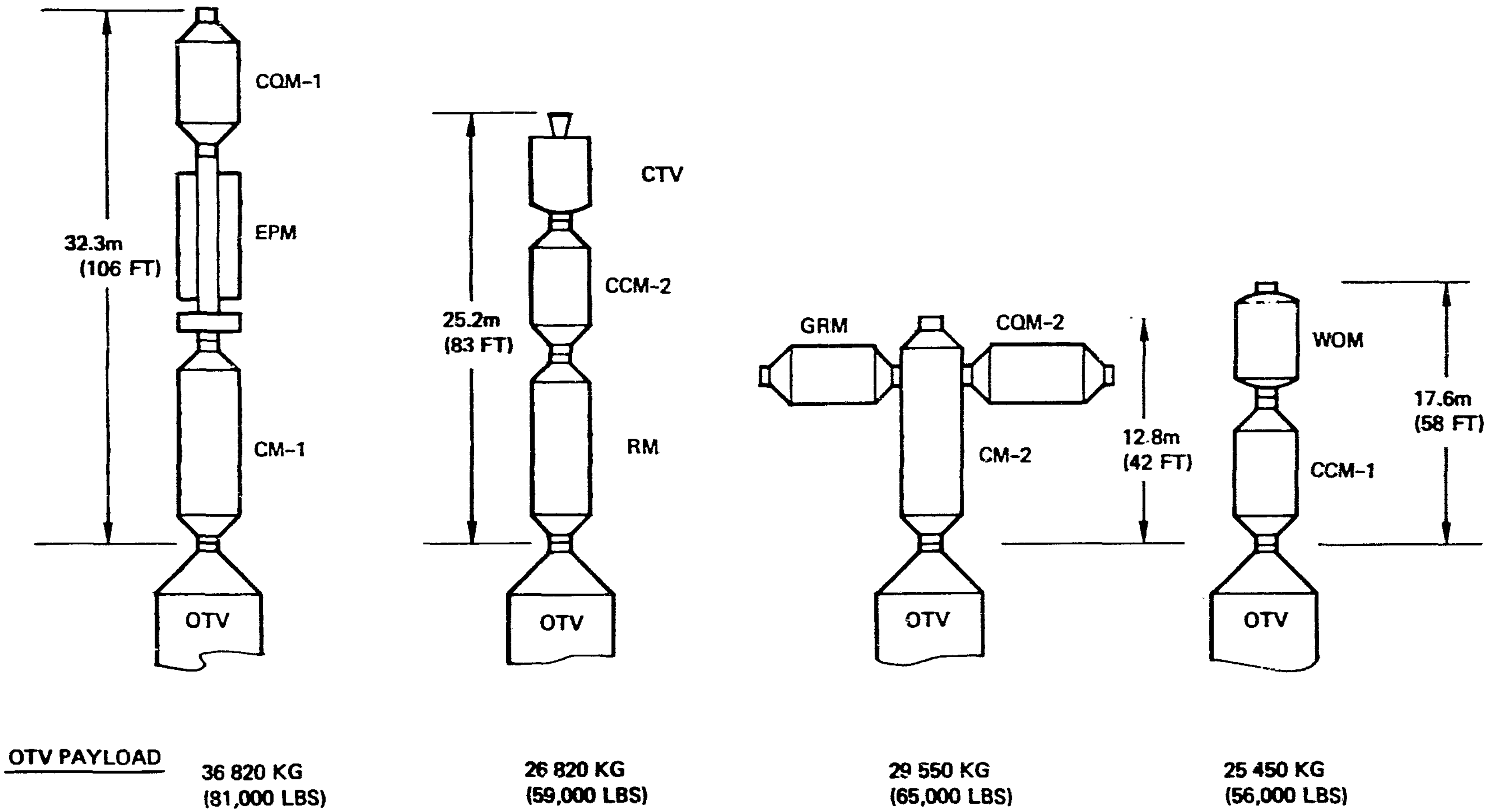


Figure 3.2-7. Modular GSS Delivery—Four Flights

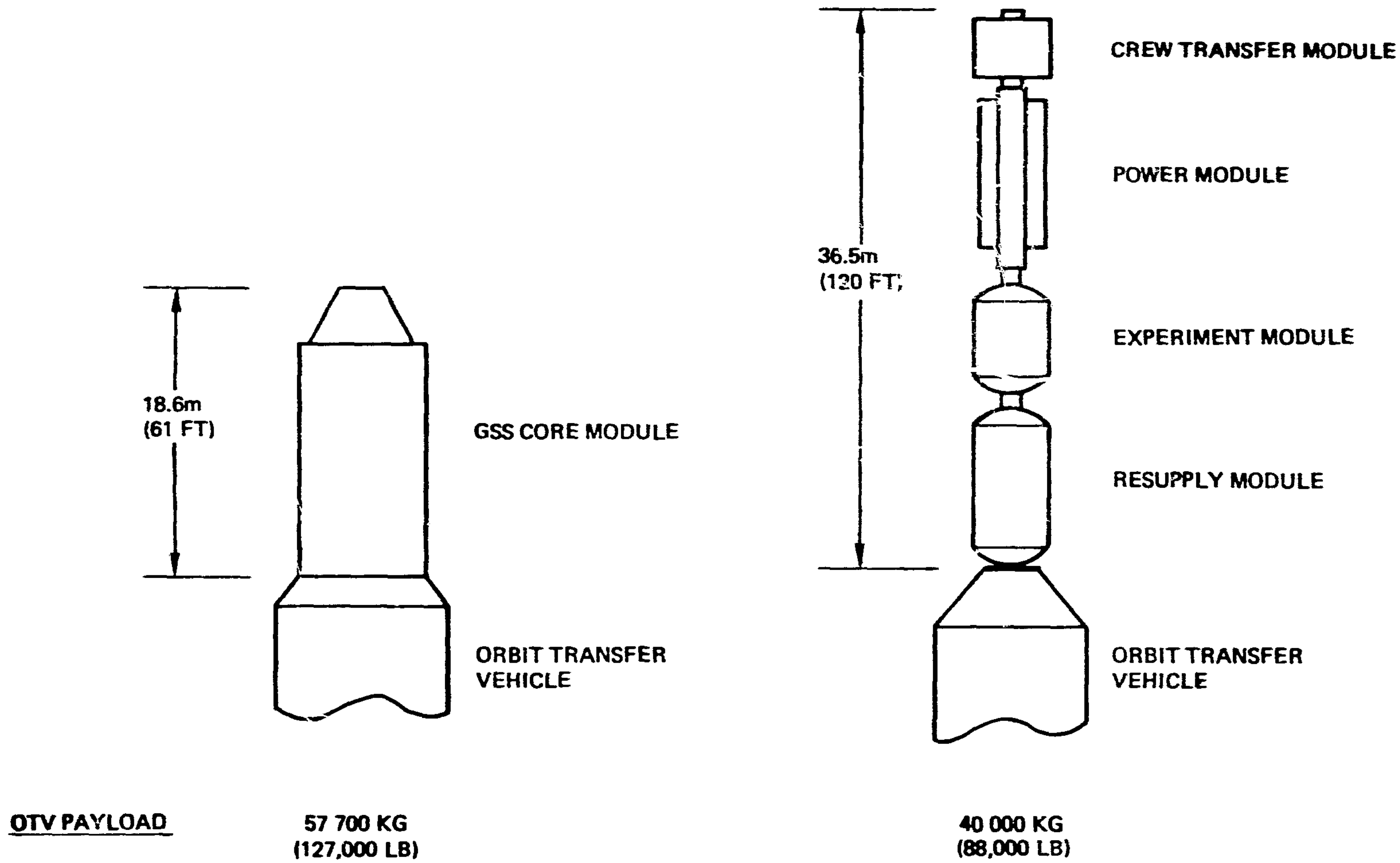
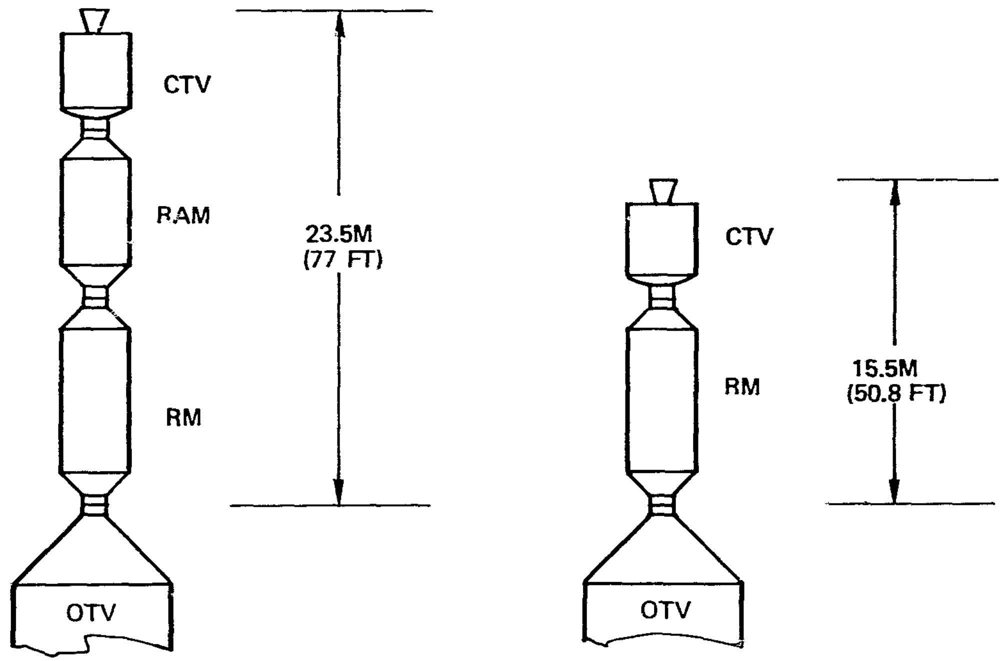


Figure 3.2-8. Unitary GSS Delivery—Two Flights

6 MONTH CREW ROTATION
AND RESUPPLY INTERVALS



OTV PAYLOAD	UP	33 200 KG (73,000 LBS)	25 000 KG (55,000 LBS)
	DOWN	15 000 KG (33,000 LBS)	15 000 KG (33,000 LBS)

Figure 3.2-9. Crew/Resupply/ASM Delivery

3.2.1.3.2.3 Operational Constraints

No operational constraints were found.

3.2.1.4 Mission/Transportation Modes and Operations

3.2.1.4.1 Transportation Options

Transportation requirements include Earth launch and orbit transfer. Earth launch options are the space shuttle and a shuttle-derived HLLV.

The principal OTV transportation candidates considered for the GSS mission are as follows:

- LO_2/LH_2 reusable single stage system (selected as representative system)
- LO_2/LH_2 one and one-half stage system that has a reusable main stage and expendable drop tanks.
- LO_2/LH_2 two stage system that has an expendable delivery stage and reusable return stage.
- LO_2/LH_2 common stage system consisting of two equal size systems, both reusable.
- LO_2/MMH common stage system consisting of two equal size systems, both reusable.
- LH_2 Nuclear Hydrogen-heater (Nerva type) reactor.
- Nuclear-Electric Tug (NET).
- Solar-Thermal Electric Propulsion System (STEPS) tug.

3.2.1.4.2 Representative Transportation Mode and System

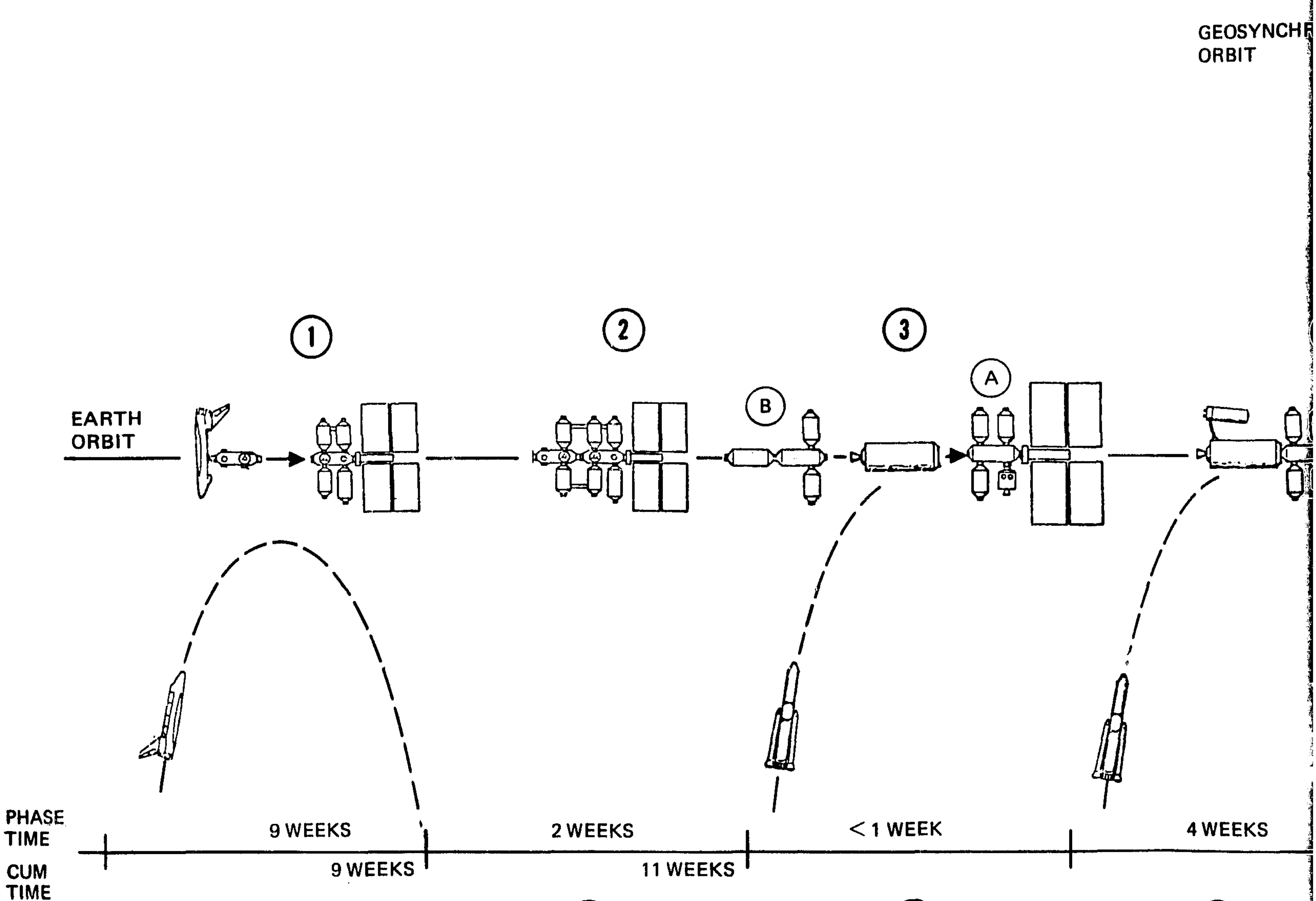
The representative transportation system includes the space shuttle and a heavy-lift space shuttle for Earth launch of the mission and transportation elements, and a large single-stage LO_2/LH_2 orbit transfer vehicle for transportation between low Earth orbit and geosynchronous orbit. The basic station is assembled in Earth orbit and then split into two subassemblies for transfer to geosynchronous orbit. The resupply mission includes one CTV and one RM.

3.2.1.4.2.1 Transportation Sequence

The reference sequence and operations associated with the GSS mission are illustrated in figure 3.2-10 using a LO_2/LH_2 single stage OTV and both the shuttle and HLLV for Earth launch. The principal features of this mode are as follows:

- A modular station is used with single modules and CTV's are delivered to low Earth orbit (LEO) by the space shuttle (SS).

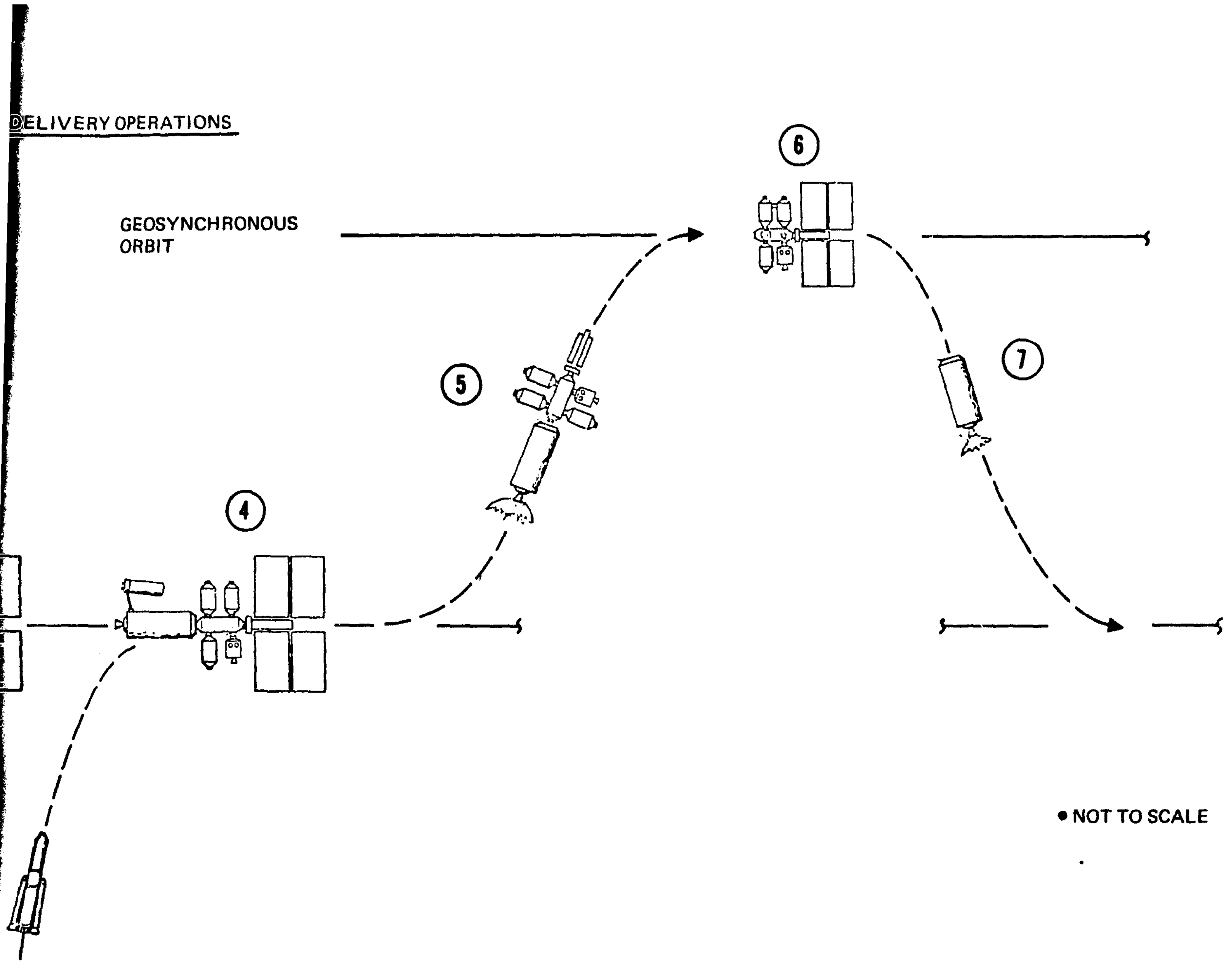
STATION DELIVERY OPERATIONS



1	2	3	4
• DELIVER 9 STATION MODULES AND ONE CTV (4 CREWMEN) TO ORBIT WITH SS . • 9 FLIGHTS	• CHECKOUT ASSEMBLED STATION	• SEPARATE STATION INTO TWO CLUSTERS (A AND B) • DELIVER OTV WITH HLV • DOCK OTV WITH CLUSTER A	• DELIVER OTV TAN WITH HLV-2 FLIGH • FUEL OTV • DISPOSE OF TANKE

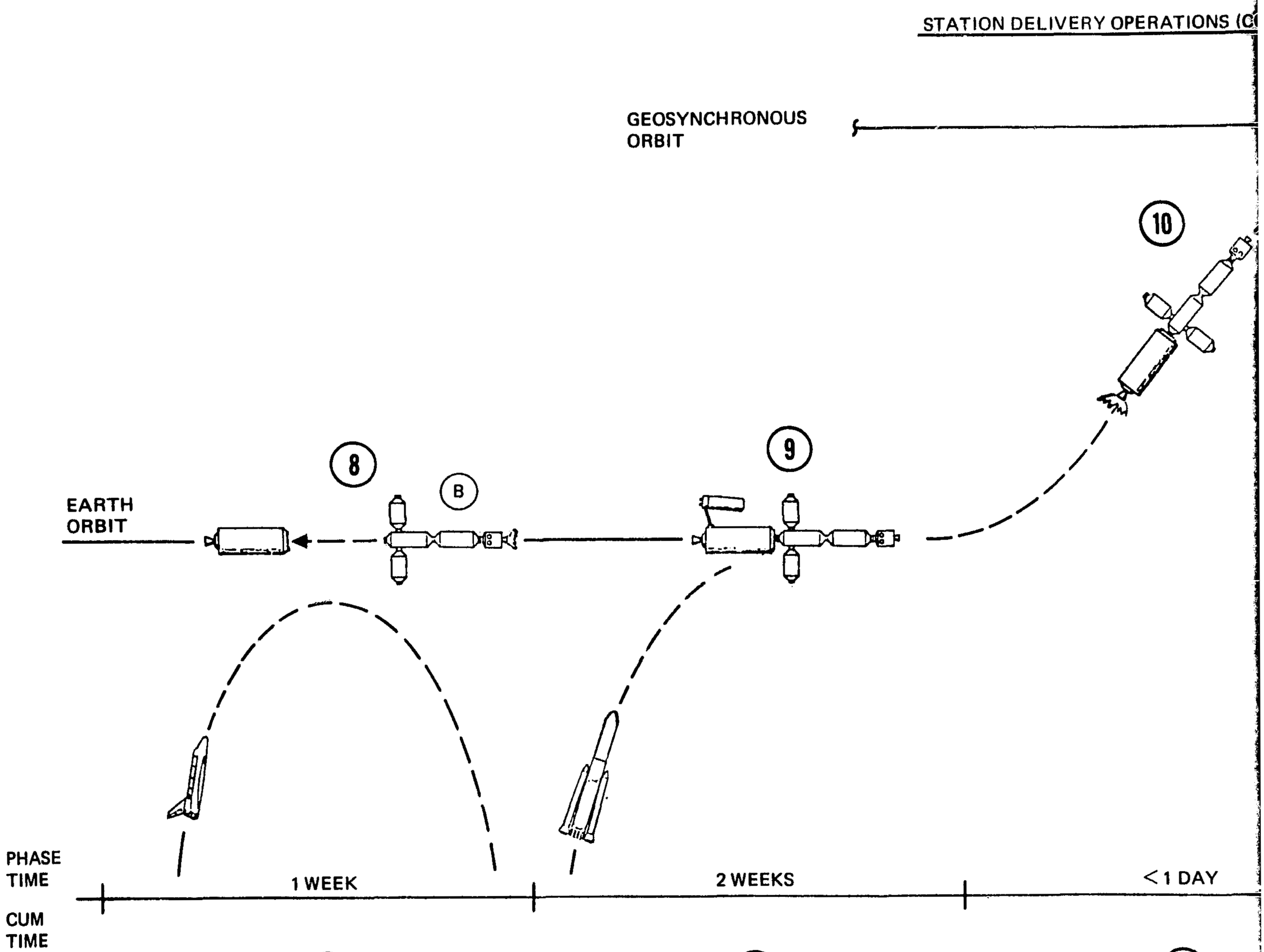
FOLDOUT FRAME

/



4 WEEKS	< 1 DAY	< 1 WEEK	< 1 DAY
		17 WEEKS	
4	5	6	7
• DELIVER OTV TANKERS WITH HLV-2 FLIGHTS • FUEL OTV • DISPOSE OF TANKERS	• STOW SOLAR ARRAY • DELIVER CLUSTER (A) TO GEO WITH OTV	• DEPLOY SOLAR ARRAY • ACTIVATE ALL SUB SYSTEMS • CHECKOUT	• RETURN OTV TO EARTH ORBIT

Figure 3.2-10 GSS Mission Transportation Sequence (Sheet 1)

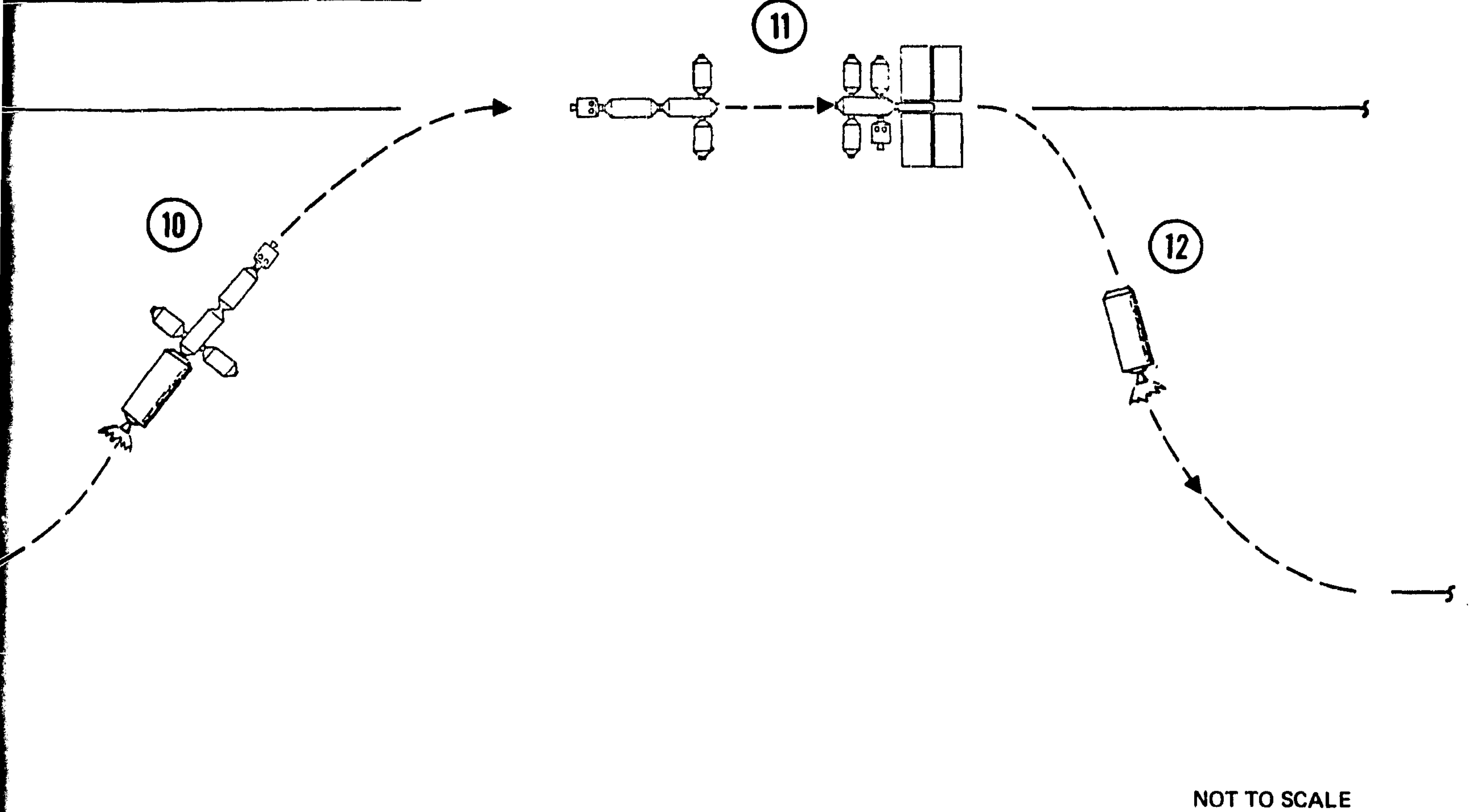


8	9	10
• DELIVER 2ND CTV (4 CREWMEN) WITH STS • DOCK CLUSTER B TO OTV WITH CTV	• DELIVER OTV TANKERS WITH HLV • FUEL OTV • DISPOSE OF TANKERS	• DELIVER CLUSTER TO GEO WITH OTV

FOLDOUT FRAME

1

DELIVERY OPERATIONS (CONTINUED)

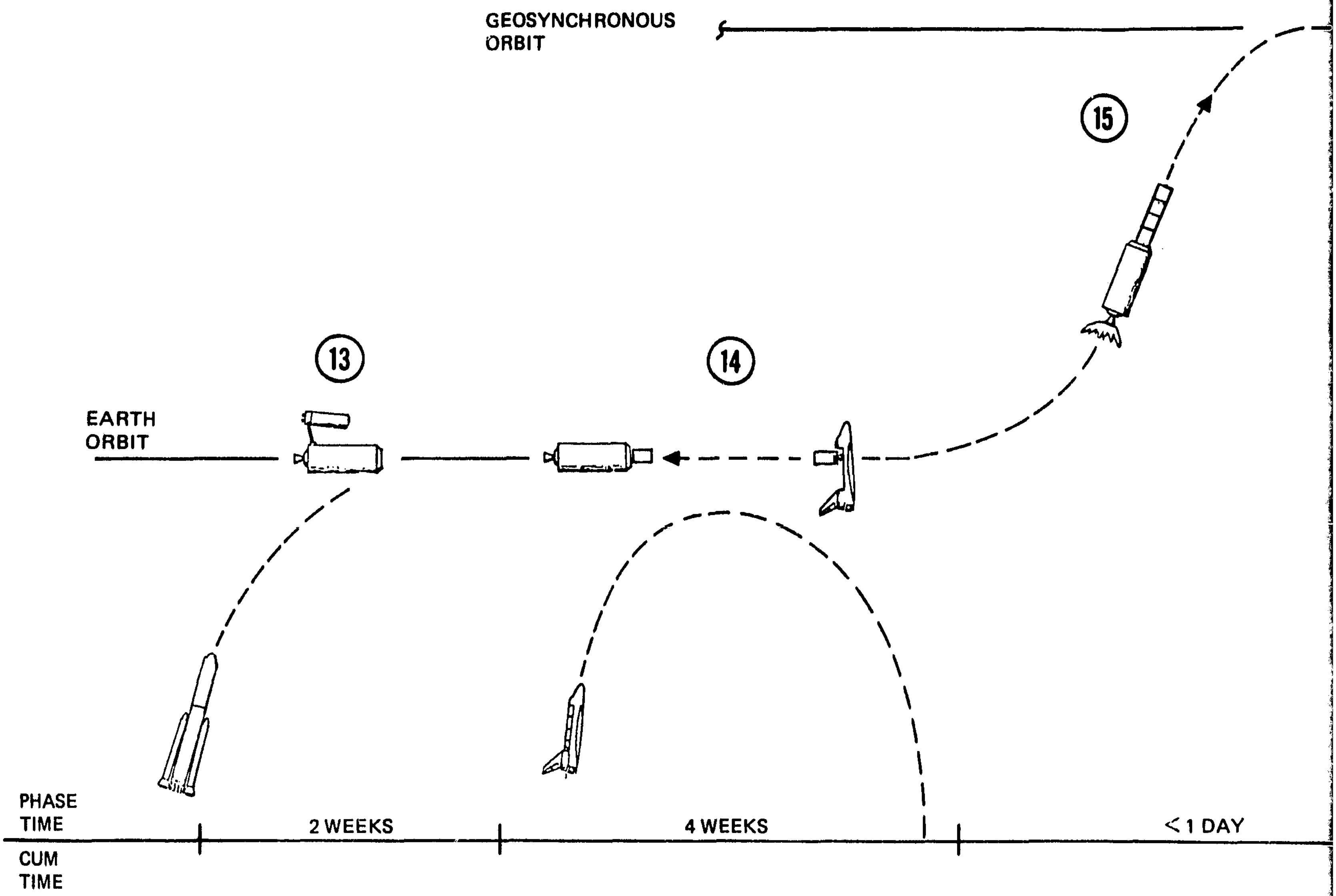


< 1 DAY	< 1 WEEK	< 1 DAY
10	11	12
• DELIVER CLUSTER (B) TO GEO WITH OTV	• SEPARATE CLUSTER (B) FROM OTV • DOCK CLUSTER (B) TO CLUSTER (A) WITH CTV • CHECKOUT ASSEMBLED STATION	• RETURN OTV TO EARTH ORBIT

Figure 3.2-10 GSS Mission Transportation Sequence (Sheet 2)

FOLDOUT FRAME 2

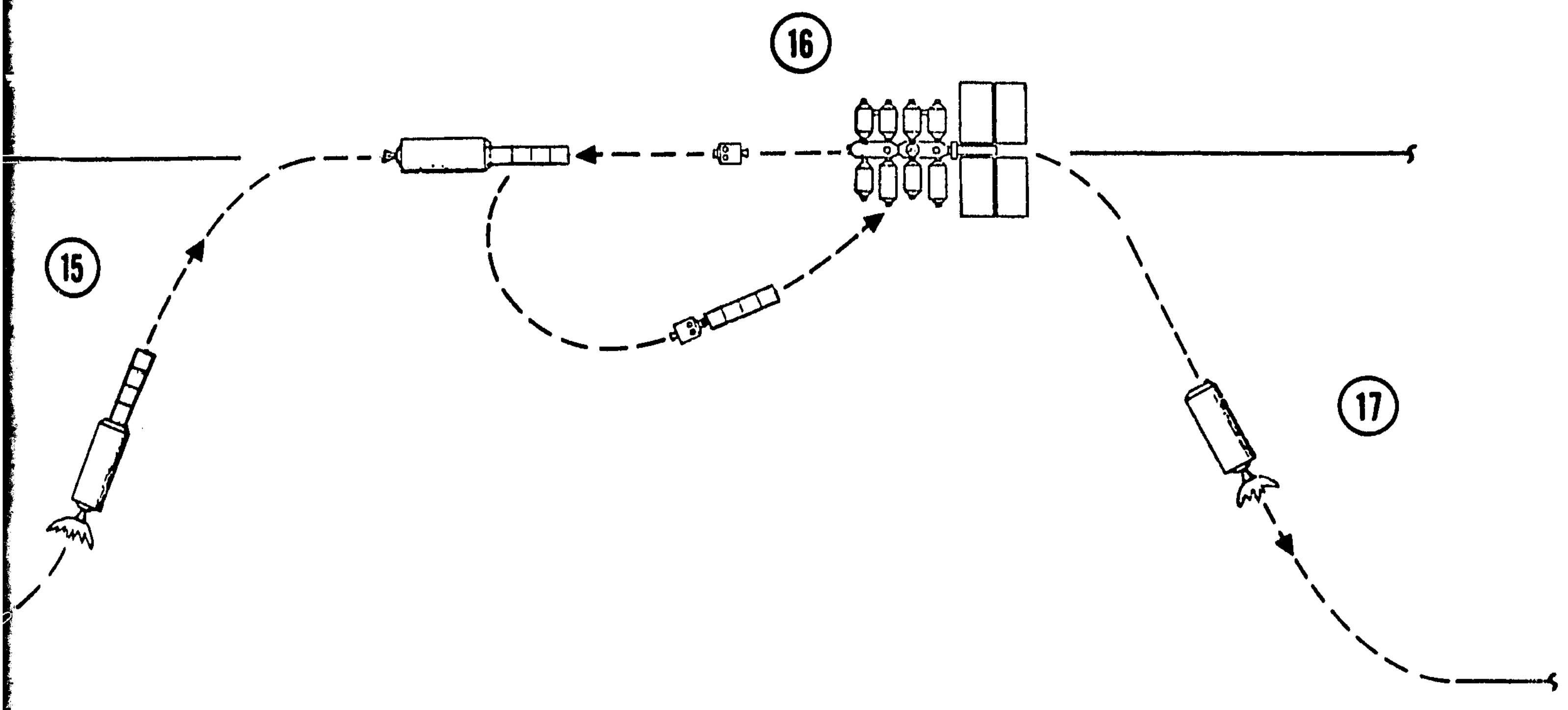
APPLICATION AND SCIENCE MODULE (ASM) DEL



13	14	15
• DELIVER OTV TANKERS WITH HLV • FUEL OTV • DISPOSE OF TANKERS	• DELIVER 3-4 ASM'S AND DOCK TO OTV USING SS • TOP OFF OTV PROPELLANT	• DELIVER ASM'S TO GEO WITH OTV

FOLDOUT FRAME

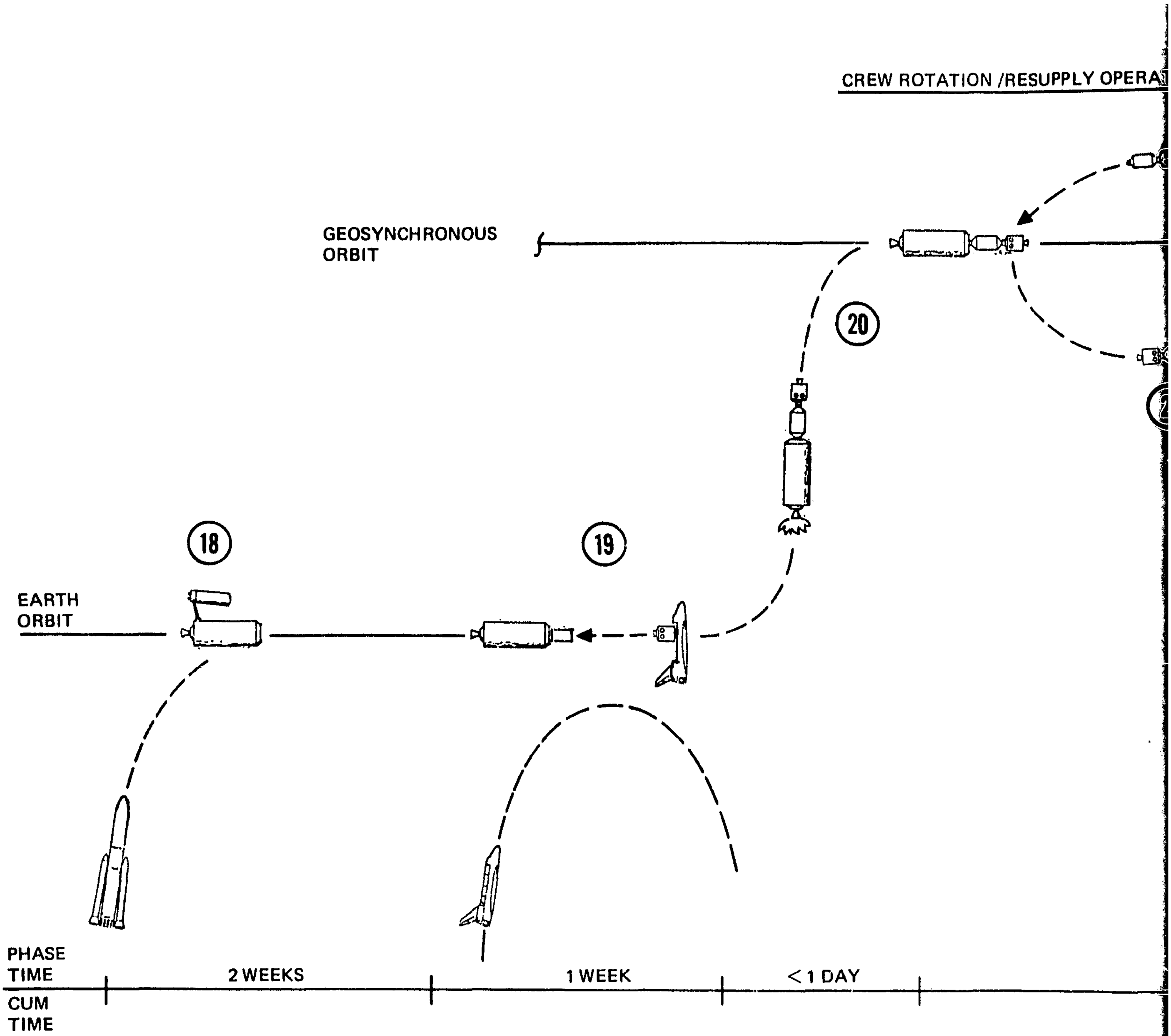
SCIENCE MODULE (ASM) DELIVERY OPERATIONS



< 1 DAY	< 1 WEEK	< 1 DAY
	7 WEEKS	

15	16	17
• DELIVER ASM'S TO GEO WITH OTV	• UNDOCK CTV FROM STATION • DOCK CTV TO ASM'S AND UNDOCK FROM OTV • DOCK ASM'S TO STATION USING CTV	• RETURN OTV TO EARTH ORBIT

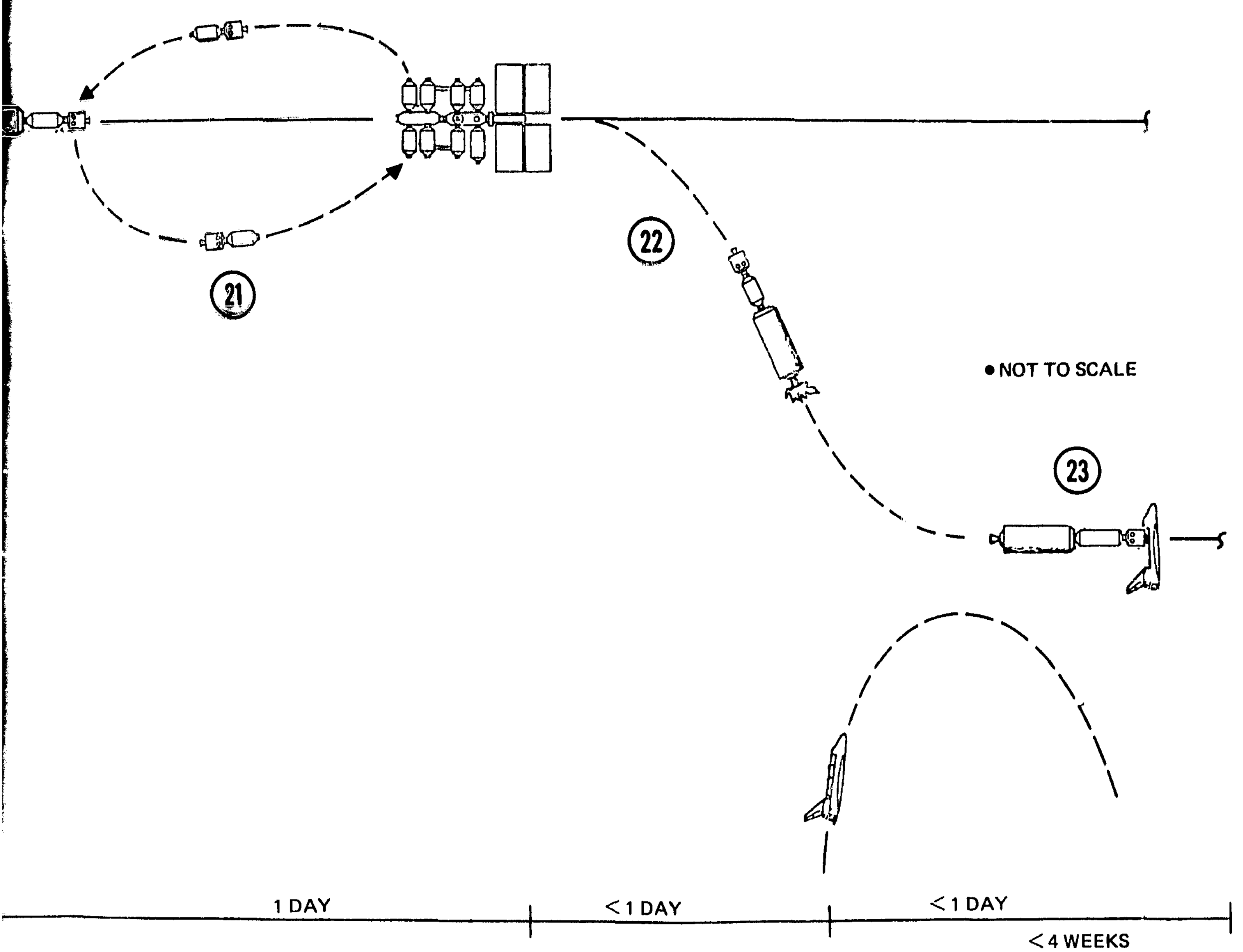
Figure 3 .2- 10 GSS Mission Transportation Sequence (Sheet 3)



18	19	20	
• DELIVER OTV TANKERS WITH HLV • FUEL OTV • DISPOSE OF TANKERS	• DELIVER NEW CTV (8 CREWMEN) AND RESUPPLY MODULE (RM) WITH STS • TOP OFF OTV PROPELLANT	• DELIVER CTV/ RM TO GEO WITH OTV	• UNDOCK NEW • DOCK RM TO S • UNDOCK OLD STATION AND

FOLDOUT FRAME /

ION /RESUPPLY OPERATIONS



21	22	23
• UNDOCK NEW CTV/RM FROM OTV • DOCK RM TO STATION USING CTV • UNDOCK OLD CTV/RM FROM STATION AND DOCK TO OTV	• RETURN CTV (8 CREWMEN) AND RM TO EARTH ORBIT WITH OTV	• RETRIEVE CTV/RM FROM ORBIT AND RETURN TO EARTH USING SS

Figure 3.2-10 GSS Mission Transportation Sequence (Sheet 4)

FOLDOUT FRAME

- A large LO_2/LH_2 single stage OTV is delivered to LEO using a HLLV and is connected with one of the station clusters.
- Fueling of the OTV is completed through use of a tanker delivered by a HLLV.
- The OTV delivers one station cluster to geosynchronous Earth orbit (GEO), returns to LEO and connects with the other station cluster.
- The OTV is again fueled and delivers the second station cluster to GEO for assembly with the first cluster and the OTV returns to LEO for future use.
- Application and science modules (ASM) are delivered to LEO with the SS and delivered to GEO with the OTV.
- At GEO, payloads (2nd station cluster, RM, ASM) delivered by the OTV are transported to the station by a CTV.
- Crew rotation/resupply occurs at 6 month intervals with a previous CTV and expended RM returned to LEO using the OTV.
- Recovery of the CTV (crew) and RM in LEO is accomplished with the SS.

A mission history including elapsed time, ΔV and mass remaining is presented in table 3.2-11.

3.2.1.4.2.2 **Transportation Sizing**

Orbit Transfer Vehicle--The most desirable size for a given type of OTV is one that satisfies all anticipated delivery requirements associated with the mission. For the GSS mission the most significant delivery requirements are those associated with placement of the station and the periodic crew rotation/resupply.

The process for obtaining the most desirable OTV size employed parametric performance maps in terms of delivery and return payload capability for each OTV candidate for this mission. Superimposed on the performance maps were the specific delivery and return payload requirements associated with the various options. The resulting plot (Figure 3.2-11) shows the match-up between transportation systems and mission requirements. An OTV mass of approximately 250 000 kg (550,000 lbs) satisfies the delivery of the modular station if delivered in two separate clusters and the requirements of crew rotation/resupply. Delivery of a unitary station in two sections can also be satisfied.

Table 3.2-11. GSS Mission History — Single Stage LO_2/LH_2 OTV

Event	Elapsed Time	Δ V		Mass Remaining	
	Hr	MPS	FPS	KG	LB
Initial mass				274,200	604,500
Ascent injection	.5	2,615	8,579	152,000	335,100
Circularize	5.5	1,837	6,026	100,300	221,000
Separate payload 25,000 kg (55,100 lb)	6.5			75,300	165,900
Return payload 15,000 kg (33,100 lb)	28			90,300	199,000
Descent injection	29.7	1,830	6,003	59,700	131,600
Earth orbit insertion		2,475	8,119	34,000	75,100
(OTV inert)				(19,000)	(42,000)
(Payload)				(15,000)	(33,100)

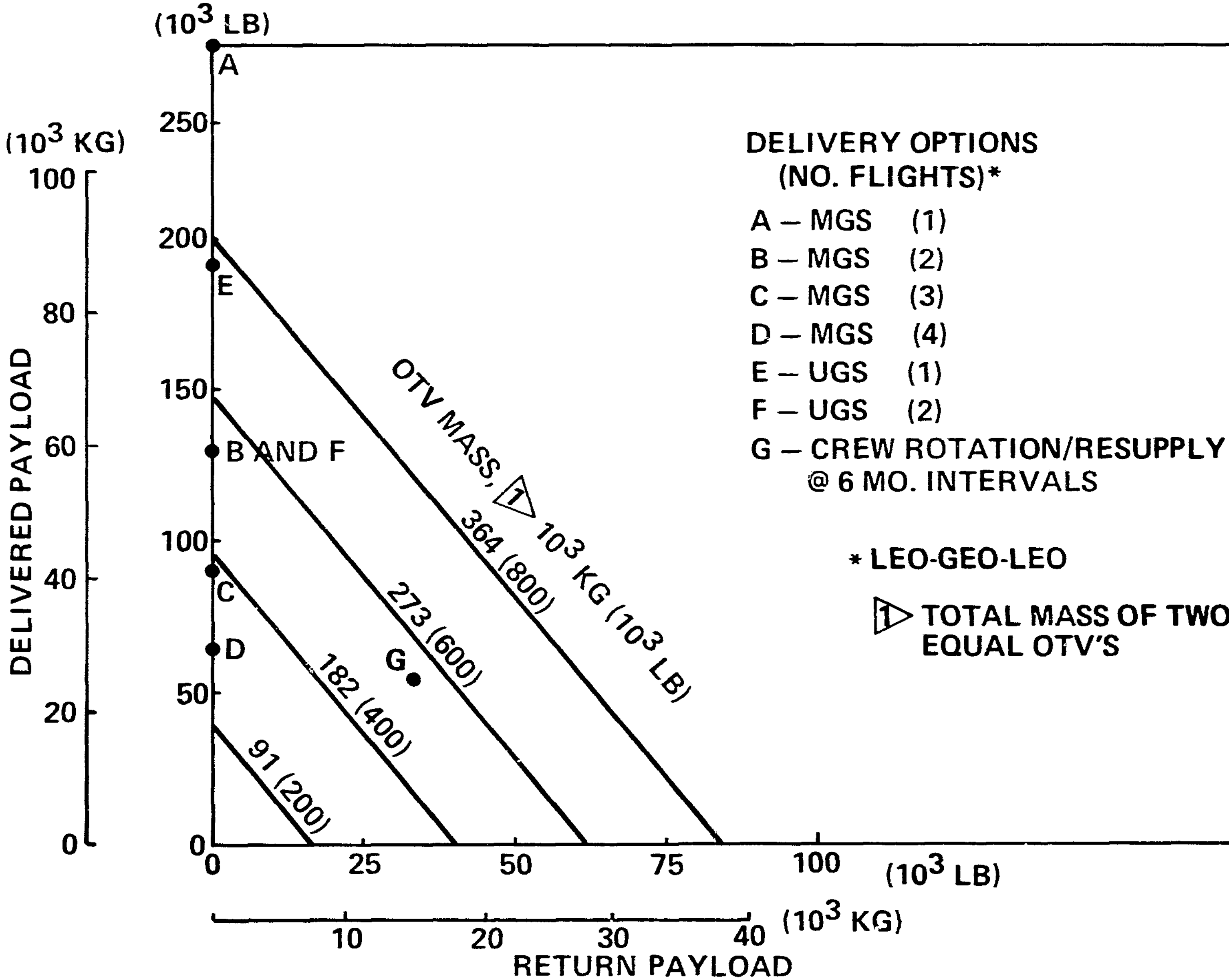


Figure 3.2-11. Single Stage LO_2/LH_2 OTV Capability for GSS

Selection of an OTV size that delivers the stations in any combination other than two sections will therefore result in over or undersizing relative to that required for crew rotation/supply.

3.2.1.4.2.3 Operational Factors

Abort–Crew return to Earth from the station in the event of an emergency is required. Four options are available:

1. A rescue, to the station and returning the crew to Earth using either a single launch from Earth or a launch staged from a low-orbit space station is the reference option. To support this option, the CTV is used as an emergency shelter.

2. A direct entry module similar to the Apollo CM is another option. Considering a mission longitude of 100°W as typical, an extra pass in the 10.5-hour elliptic descent orbit followed by a skip entry will allow recovery near the continguous U.S. The delta V requirement is 1 915 m/sec (6,300 ft/sec).

3. An aerobraking/entry glider (see paragraph 3.2.2).

4. A propulsive return to low Earth orbit with a crew transport module, requiring 4 200 m/sec (14,000 ft/sec) is a fourth option.

Mission Profiles and Rendezvous Techniques–Nominal mission profiles require transfers between low orbits and synchronous orbit including longitude phasing. Geosynchronous orbits have an orbit period equal to the Earth's rotation period. The payloads described are therefore stationary over a selected Earth longitude. Phasing is done by a combination of wait in low orbit (typically 25 degrees longitude per orbit), extra passes in the 10.5-hour elliptic transfer orbit (about 160° per orbit), and a final phasing orbit with apogee at geosynchronous orbit and a period slightly less than the 23.93-hour geosynchronous period. The final phasing orbit is adjusted to provide the last increment of phasing not conveniently obtained from the prior phasing. These maneuvers require coast time; this results in a total transfer time on the order of 40 hours depending on specific longitude desired and low orbit parameters. The maneuvers do not incur delta V penalties. They were described in the context of an up transfer; analogous maneuvers are required for the down transfer if return to a particular (e.g., space station) low orbit is required.

Rendezvous in the low Earth orbit is required both for the space shuttle and the returning OTV. The rendezvous technique employs circular coplanar concentric phasing orbits with the active element at the lower altitude. Rendezvous in the geosynchronous orbit occurs at the termination of

the final phasing (elliptic) orbit. The rendezvous itself takes place in a fraction of an orbit and is analogous to terminal rendezvous maneuvers in a low orbit.

Crew Involvement and Timelines—Mission and operations crews will be involved in all phases of the GSS mission. Crew involvement simplifies transportation operations such as rendezvous, docking, and orbital assembly since the crew can be relied upon as the principal control element. Crew timelines do not have any identified effect on transportation requirements. Crew rotation requirements were described in Paragraph 3.2.1.2.3.

Control Functions and Requirements—Geosynchronous round trip missions require precision guidance and navigation control to accomplish the transfers and achieve rendezvous phasing orbits within nominal performance allocations. Control functions include derivation and execution of steering commands, measurement and control of delivered impulse, and propellant management. Autonomous onboard control is highly desirable.

Network Support—Network support may be needed for orbit determination to support final rendezvous.

Geosynchronous manned stations will nominally be in continuous direct line-of-sight communications with the U.S. Large imaging applications or science modules are expected to generate very high data rates in the range 100 to 1000 megabits/sec. These will require X or Ku band links for communications to Earth.

3.2.1.4.2.4 Earth Launch Requirements Summary

The representative sequence requires 6 HLLV launches to deliver the OTV and tankers to effect station delivery, and 2 shuttle launches and 4 HLLV tanker launches for each 6-month crew rotation and resupply mission.

3.2.1.4.3 Transportation Options Comparison and Evaluation

3.2.1.4.3.1 Size and Performance Comparison

Sizing parametrics similar to Figure 3.2-11 are shown in Figures 3.2-12 through 3.2-15. As for the reference single-stage OTV, the crew rotation/resupply requirement matches well with delivery of the station in two flights. The 1-1/2 stage system was sized only for the nominal crew rotation/resupply requirement.

OTV mass comparison of the candidate transportation systems is presented in Figure 3.2-16. The payload used in this comparison is that of the crew rotation/resupply flight. The significantly lower

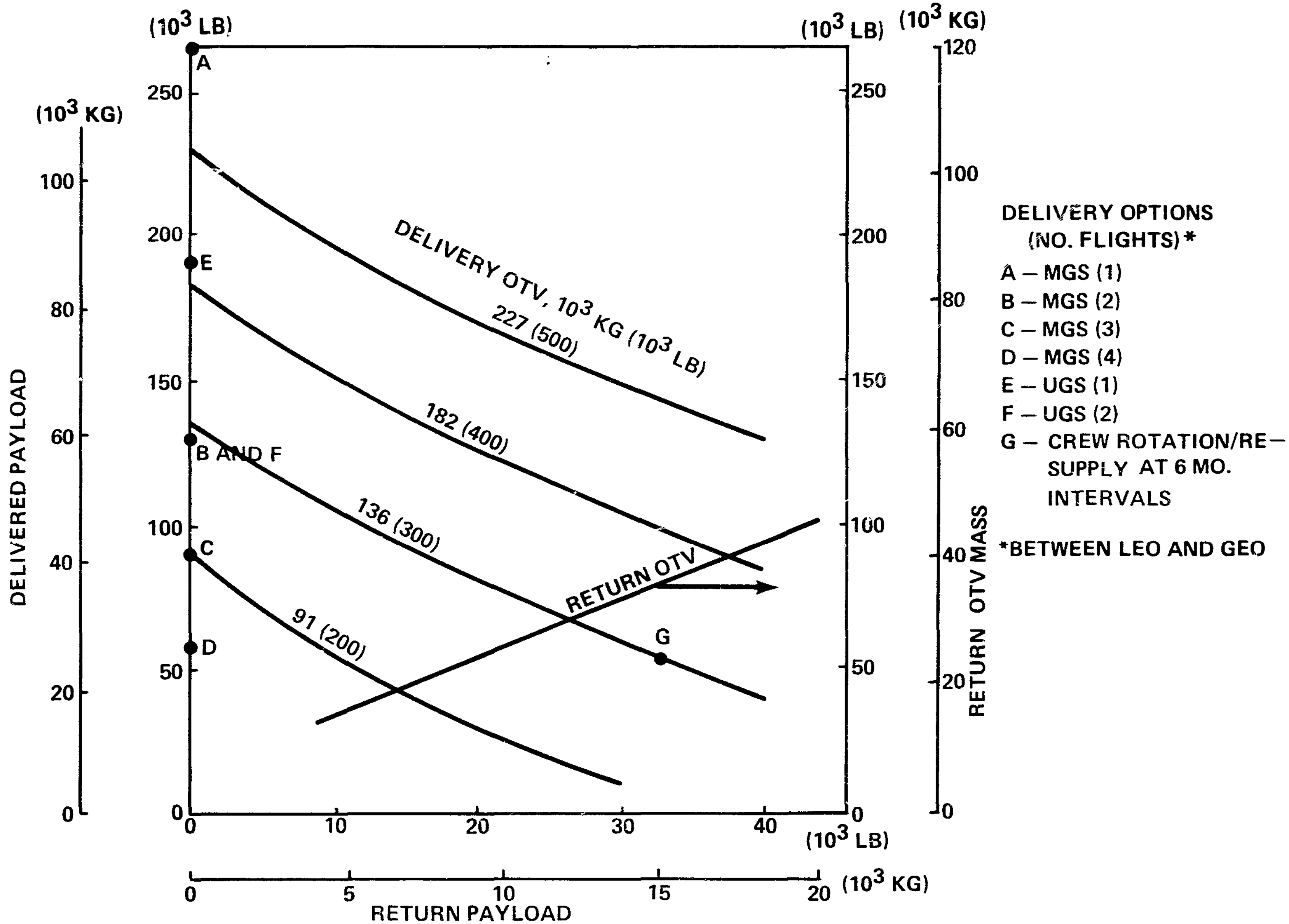


Figure 3.2-12. Two Stage LO_2/LH_2 OTV Capability for GSS

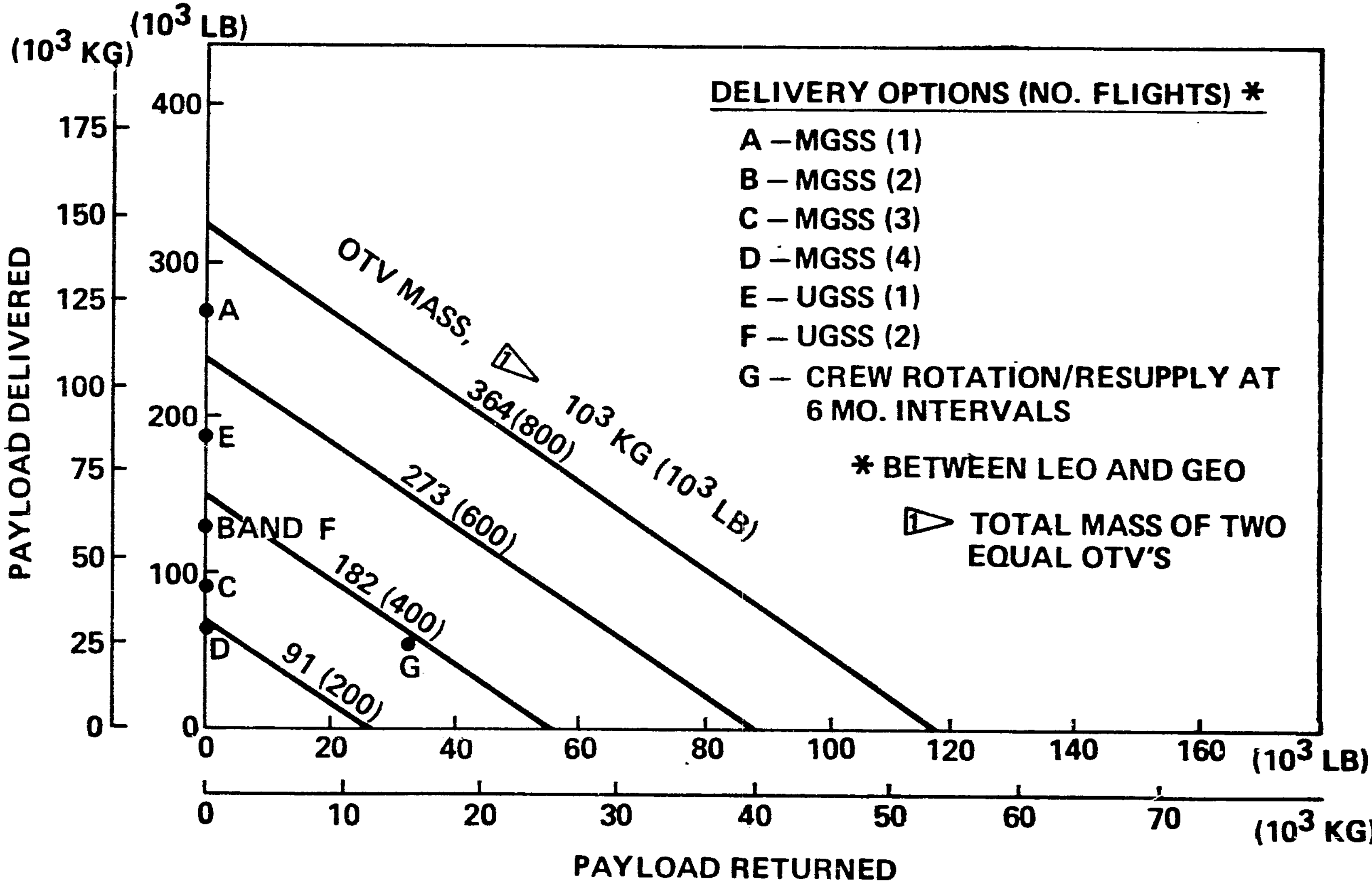


Figure 3.2-13. Common Stage LO_2/LH_2 OTV Capability for GSS

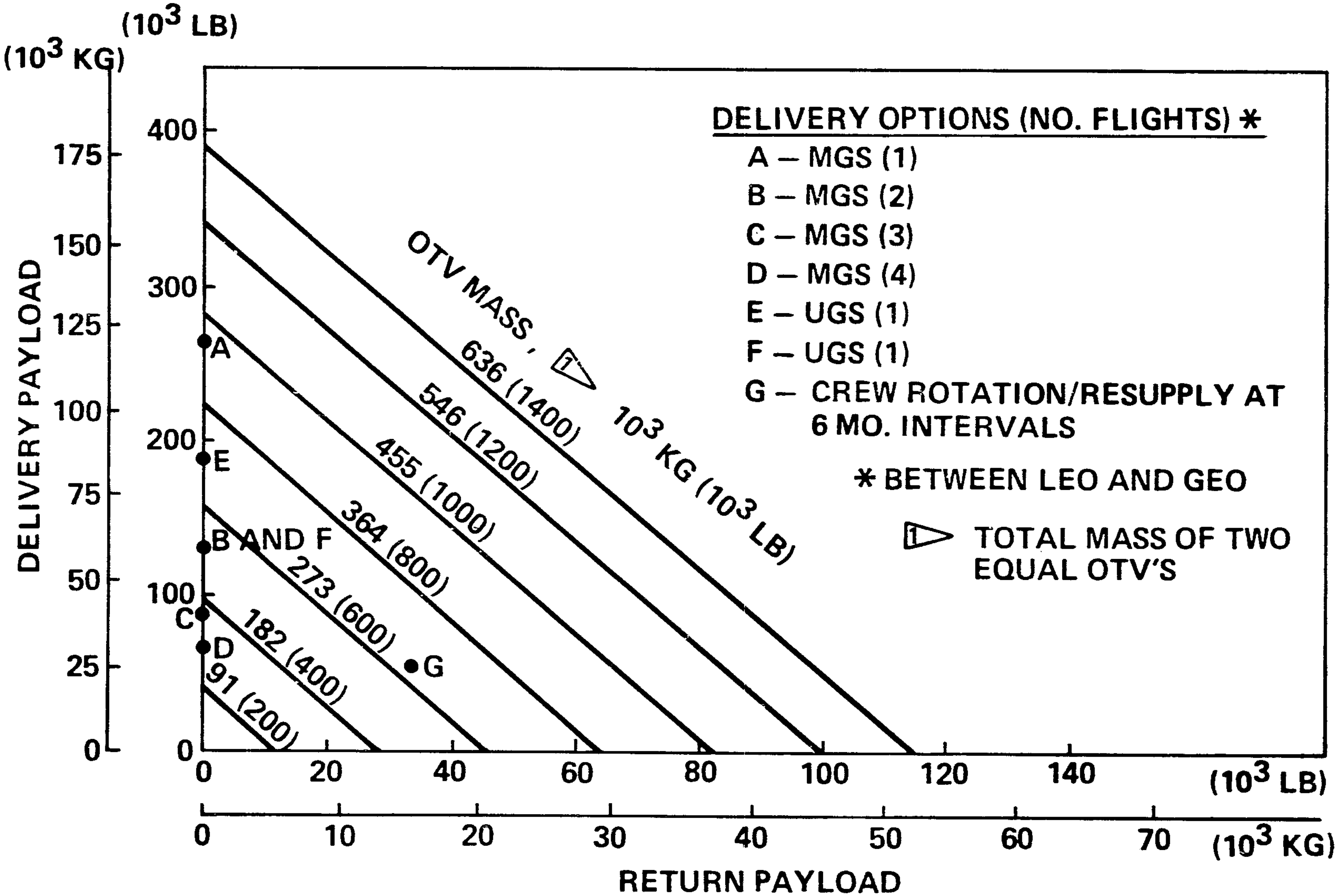


Figure 3.2-14. Common Stage LO_2/MMH OTV Capability for GSS

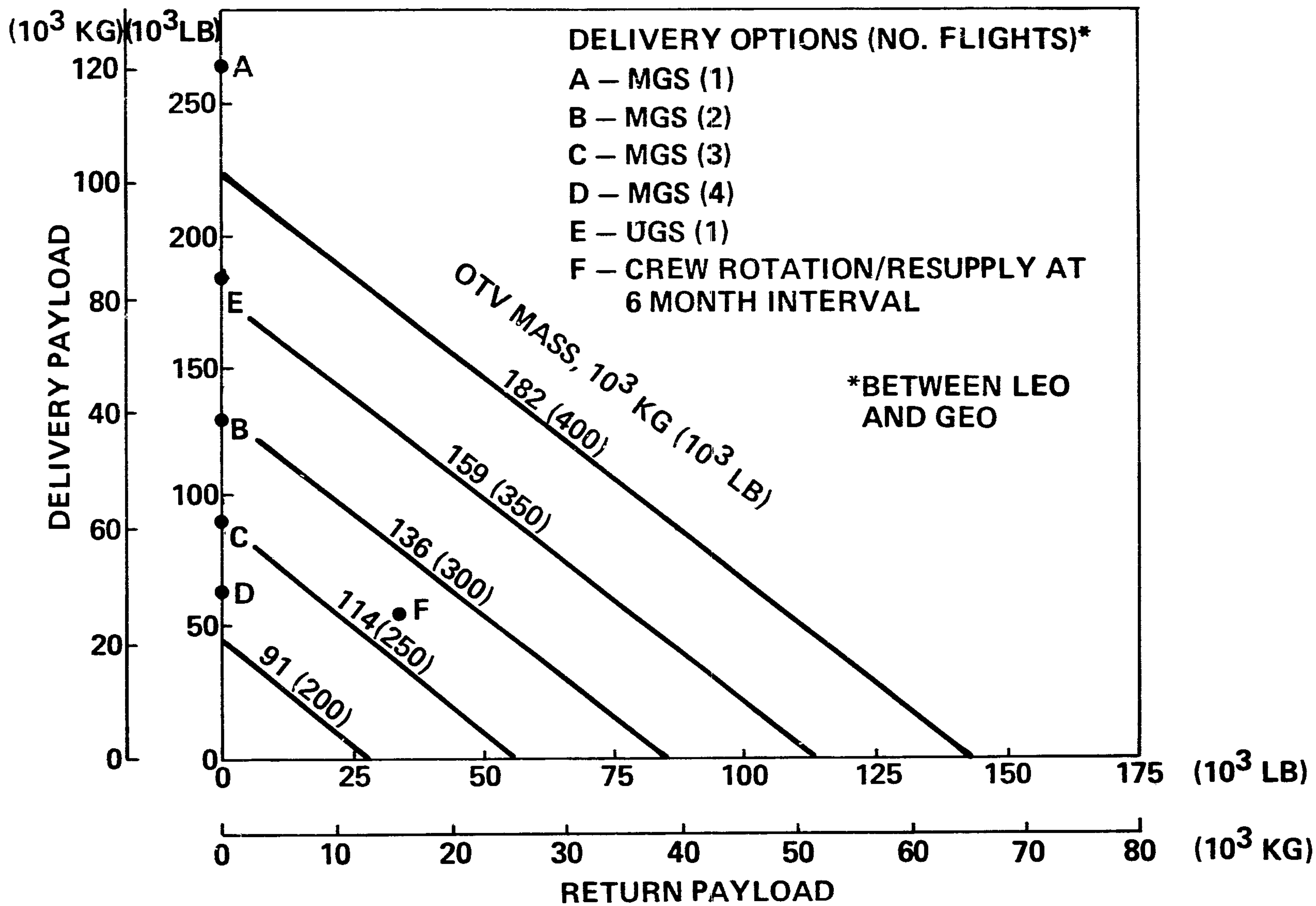


Figure 3.2-15. Nuclear Stage OTV Capability for GSS

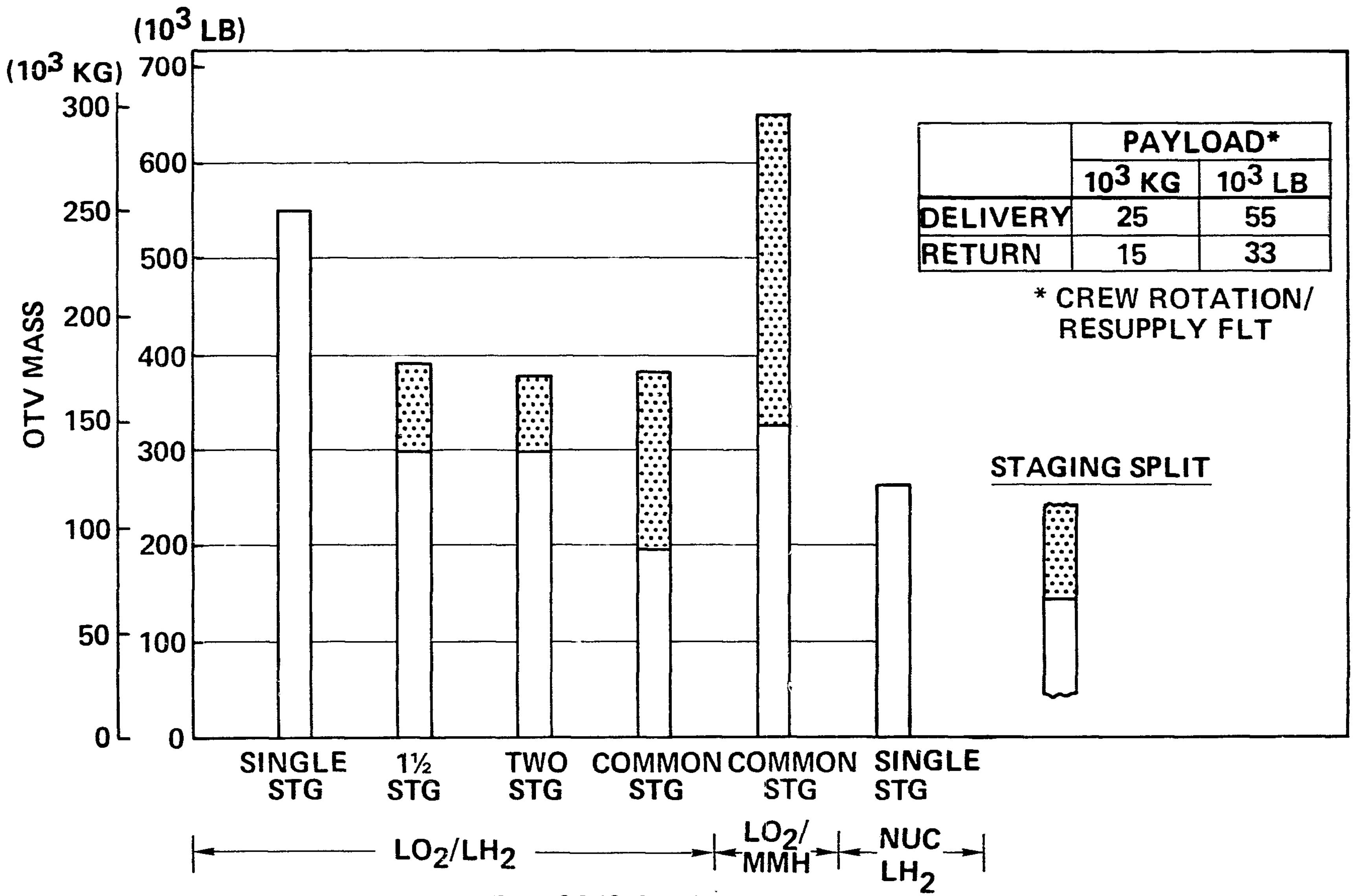


Figure 3.2-16. OTV Comparison for GSS

mass of the LO_2/LH_2 1-1/2 stage, two stage and common stage systems compared to the single stage approach is due to transporting less inerts as a result of staging. The LO_2/MMH common stage is heavier than those of the LO_2/LH_2 systems as a result of having a lower Isp (approximately 870 m/sec or 90 sec). Representative mass for nuclear electric systems is also shown. Although the total stage mass is large, the propellant mass, representing actual recurring Earth-to-orbit transportation requirements, is less than for the other alternatives. Electric systems, however, are not deemed suited for this mission (see below).

The size comparison of these systems is shown in Figure 3.2-17. Groundrules used in establishing tank lengths were 1) separate fuel and oxidizer tanks (no common bulkhead or nesting), and 2) 0.7 elliptical bulkheads. Diameters selected were those most closely associated with that necessary to have two 0.7 bulkheads form the oxygen tank since it requires the smallest volume. Electric systems are not shown; they are much larger in dimension than the vehicles shown.

Several observations can be made as a result of this comparison. The first of these is that the LO_2/LH_2 1-1/2 stage system and LO_2/MMH common stage system both are size compatible with delivery to low Earth orbit by the Space Shuttle, although off-loading of propellant will be required. The remaining stages require HLLV. Secondly, the LO_2/MMH stage is relatively small even though it has a high mass because of the high propellant density.

3.2.1.4.3.2 Earth Launch Requirements Comparison

The number of Earth launches required to delivery OTV hardware and fuel necessary to deliver the GSS elements to initiate the mission is shown in figure 3.2-18. Several of the OTV candidates are dimensionally compatible with the space shuttle. All of the OTV concepts can be launched with the HLLV with a considerable reduction in number of Earth launches. Space shuttle flights are shown with one of the HLLV options since only a portion of the HLLV capability would be required to complete the delivery of the OTV systems.

In general, for those OTV candidates that can use either launch vehicle, the HLLV requires only one-third as many launches.

Earth launches required to deliver OTV hardware and fuel necessary for the GSS crew rotation and resupply requirements are shown in figure 3.2-19. Again, the HLLV requires only one-third as many launches as the space shuttle.

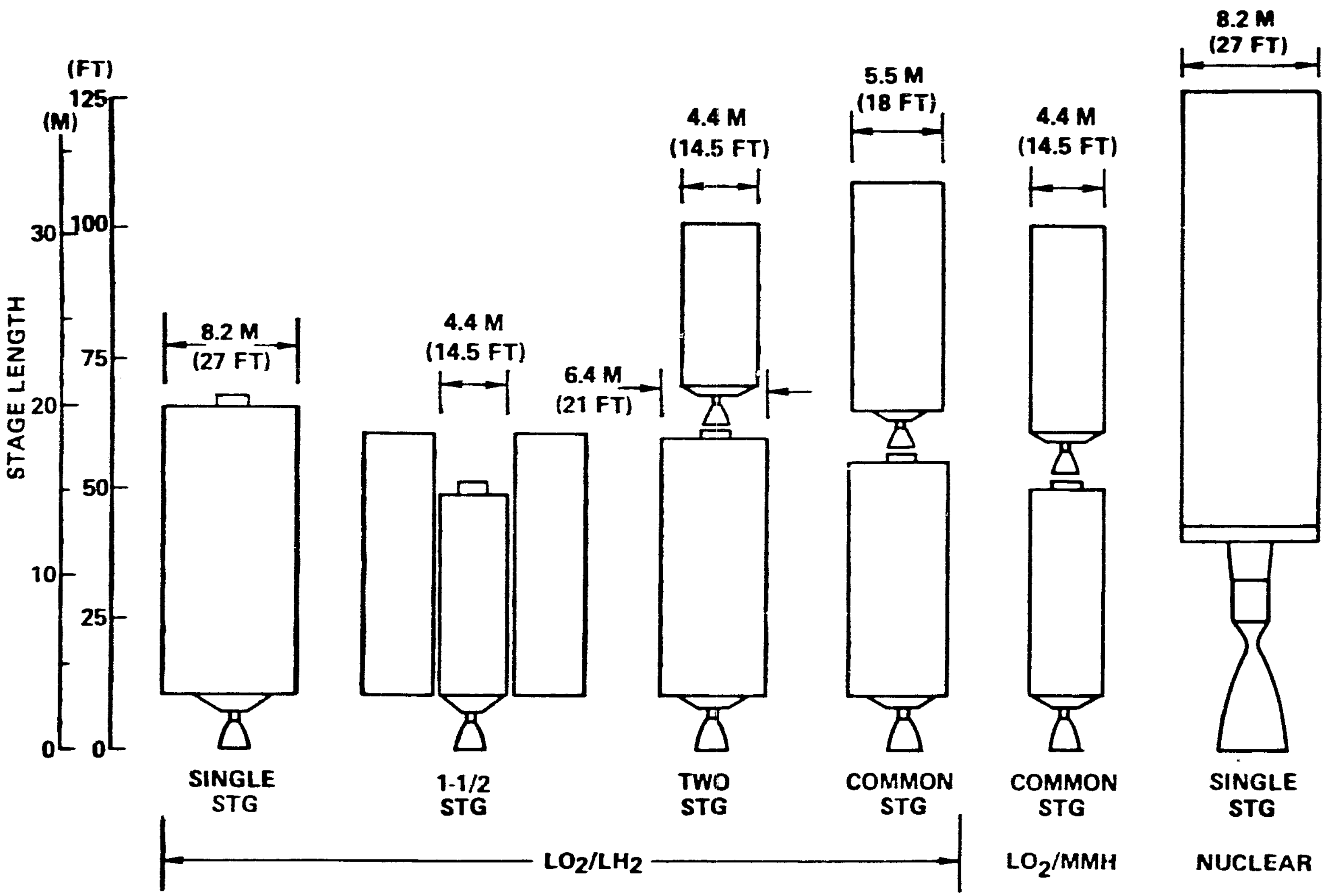


Figure 3.2-17. OTV Size Comparison for GSS

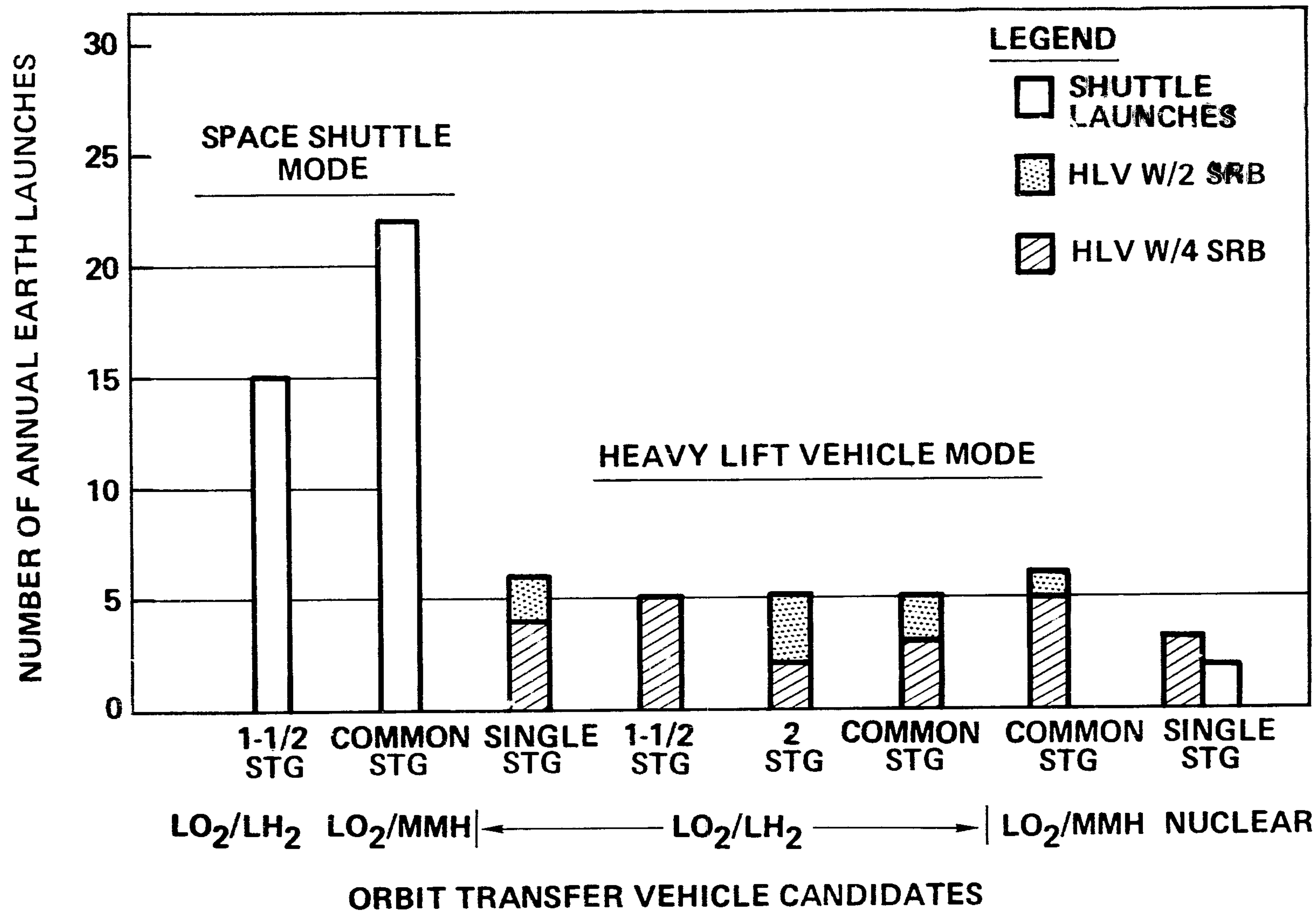


Figure 3.2-18. Earth Launches Required for GSS OTV System (Mission Start-Up)

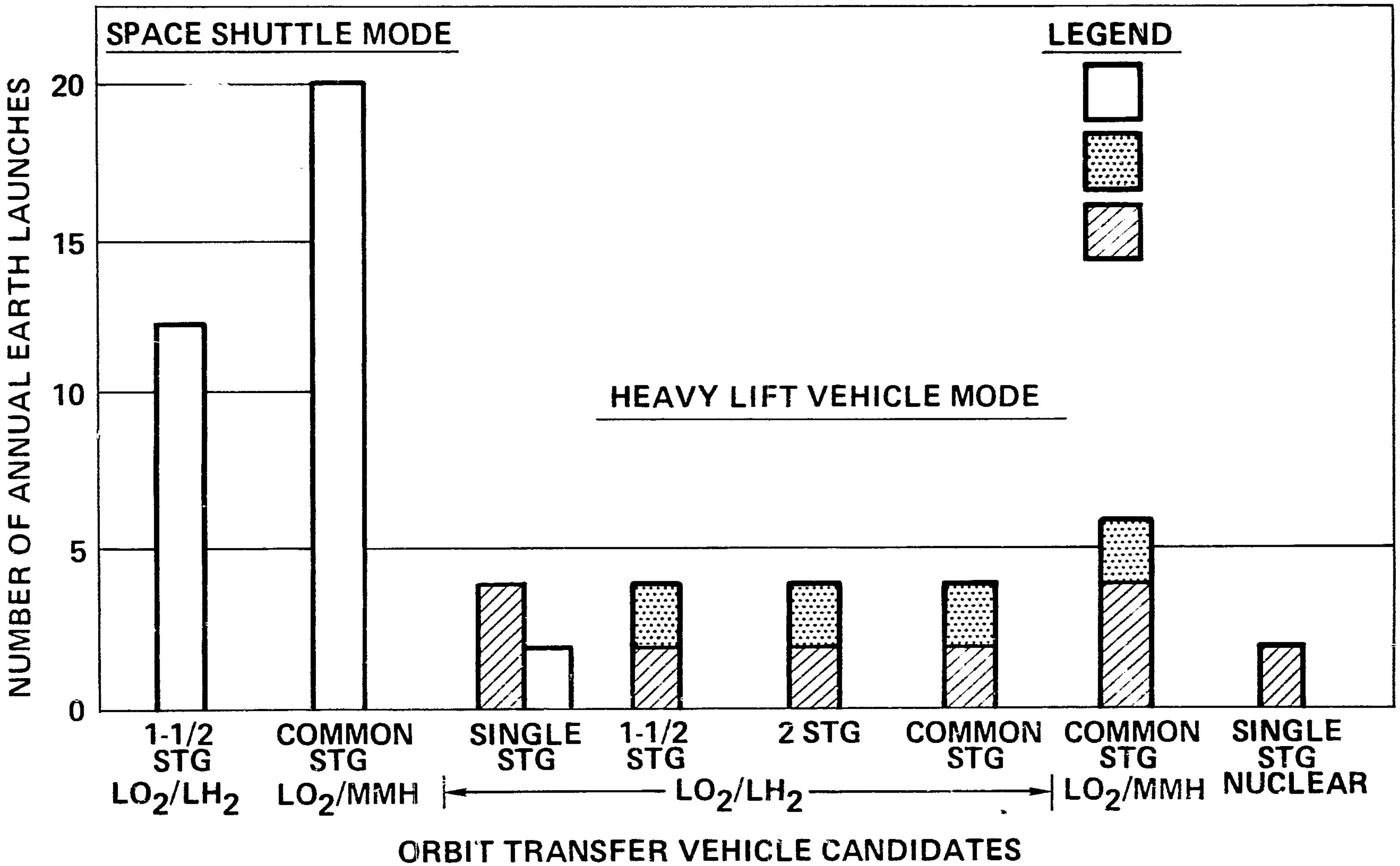


*Figure 3.2-19. Earth Launches Required for GSS OTV System (Annual Crew Rotation/Resupply)**

3.2.1.4.3.3 Operational Comparison

All the geosynchronous mission/transportation modes require precision targeting for rendezvous in geosynchronous orbit and for Earth return, either to low Earth orbit or to direct or aerobraking entry. Onboard navigation and targeting is desirable for all modes and essential for aerobraking.

In addition, all modes require some combination of docking, refueling, and recovery by the Shuttle. These requirements are summarized in Table 3.2-12

Table 3.2-12. Geosynchronous Missions Operational Factors

		TRANSPORTATION MODE \ FUNCTION	NAVIGATION & TARGETING: RENDEZVOUS	NAVIGATION & TARGETING: LOW ORBIT RETURN	NAVIGATION & TARGETING: LOW-THRUST GUIDANCE & NAVIGATION	DOCKING: HEAD-TO-HEAD	DOCKING: SIDE-BY-SIDE	DOCKING: LARGE DIAMETER INTERSTAGES	FUELING OR REFUELING	SHUTTLE RECOVERY OF HARDWARE	LOW DELTA V TUG FOR TRANSFER OPS	REACTOR DISPOSAL	ORBITAL ASSEMBLY OF VEHICLE
GEOSYNCHRONOUS MANNED STATION	SHUTTLE COMPATIBLE	1½ STAGE	●	●			●			●			
		COMMON STAGE LO_2/MMH	●	●				●	●	●			
		NUCLEAR ELECTRIC	●		●					●		●	●
		SOLAR THERMAL ELECTRIC	●		●					●			●
	REQUIRE HEAVY LIFT	SINGLE STAGE	●	●					●				
		COMMON STAGE LO_2/LH_2	●	●				●	●				
		TWO STAGE	●	●		●			●	●			
		NUCLEAR LH_2	●	●					●		●	●	

Nuclear propulsion systems present severe operational problems. The LH_2/Nerva system cannot approach either the shuttle or the manned station due to radiation hazards. A special transfer tug must be used; it approaches and departs from the LH_2/Nerva stage within a safety cone defined by the shadow shield. Fueling must be accomplished by automated or remote-controlled systems.

Spent reactor disposal is an unsolved problem. The nuclear-electric tug (NET) and solar/electric (STEPS) tug are not suited for crew rotation and resupply due to the long trip times. They could be used for station placement but represents a redundant solution since the chemical system sized for crew rotation can also do the station placement. With the NET, also, reactor disposal is a problem.

3.2.1.4.2.4 Practicality Assessment

All of the chemical modes analyzed are practical for the GSS mission. The nuclear and STEPS modes present severe operational problems and are judged impractical. Table 3.2-13 presents pro's and con's for the systems.

Table 3.2-13. Geosynchronous Transportation Mode Comparison

		ADVANTAGES					DISADVANTAGES					
		COMPATIBLE WITH SHUTTLE LAUNCH AND RECOVERY	NO STAGE-TO-STAGE DOCKING	FULLY REUSABLE	ONLY ONE STAGE TO DEVELOP	NO LH_2	RADIATION-IMPOSED OPERATING CONSTRAINTS	REACTOR DISPOSAL	MULTIPLE DOCKING TO ASSEMBLE	ORBITAL FABRICATION	SLOW TRANSFER	TWO ACTIVE VEHICLES TO TRACK
LO_2/LH_2	SINGLE STAGE		x	x	x							
LO_2/LH_2	1½ STAGE	x							x			
LO_2/LH_2	2-STAGE											x
LO_2/LH_2	COMMON STAGE			x	x							x
LO_2/MMH COMMON STAGE		x		x	x	x						x
LH_2 NUCLEAR STAGE			x	x	x		x	x				
NUCLEAR ELECTRIC TUG			x	x	x	x		x		x	x	
SOLAR ELECTRIC TUG			x	x	x	x				x	x	

3.2.2 GEOSYNCHRONOUS SATELLITE MAINTENANCE SORTIE

3.2.2.1 Mission Summary

3.2.2.1.1 General Description

Current estimates of the quantity of geosynchronous automated satellites range from 180 to over

400 by 1990. Economics associated with operating the satellites will probably necessitate repair and refurbishment rather than disposal when a failure occurs. Complexity of the satellites may prevent maintenance by automated vehicles. As a result of the above factors, the need for manned sorties to repair and refurbish automated geosynchronous satellites is a probable requirement.

The reference geosynchronous satellite maintenance sortie (GSMS) mission consists of a four man crew performing one week of maintenance operations. The mission concept is illustrated in figure 3.2-20.

The major system elements in this mission include a crew transfer vehicle (CTV) to house the four man crew for one week and the necessary repair and refurbishment provisions.

3.2.2.1.2 Mission Assumptions and Constraints

The following assumptions and constraints were used:

- 4-man crew, one week in geosynchronous orbit
- Remote manipulator maintenance system; normally no EVA
- 1 000 kg (2,200 lb) maintenance and spares payload
- Return to low Earth orbit upon severe solar flare event

3.2.2.2 Mission System Description

3.2.2.2.1 Payloads

The crew is housed in a crew transport module (CTV) very similar to that used on the lunar landing missions; see paragraph 3.3.1.2.

Mass and size characteristics of the system elements are influenced by the transportation mode used to perform the mission. One such mode is the return of the CTV back to Earth orbit upon completion of the maintenance task. Another mode is to design the manned compartment to have direct entry capability.

For the return to Earth orbit mode, the CTV is estimated to have a mass of 4 100 kg (9,040 lbs). An Apollo type direct entry vehicle would have a mass of approximately 5 930 kg (13,070 lbs).

Repair and refurbishment provision will be the same for both modes with an estimated mass of 1 000 kg (2,200 lbs).

Pickup Points—These payloads are normally launched integrated with an orbit transfer vehicle, either by space shuttle or a HLLV.

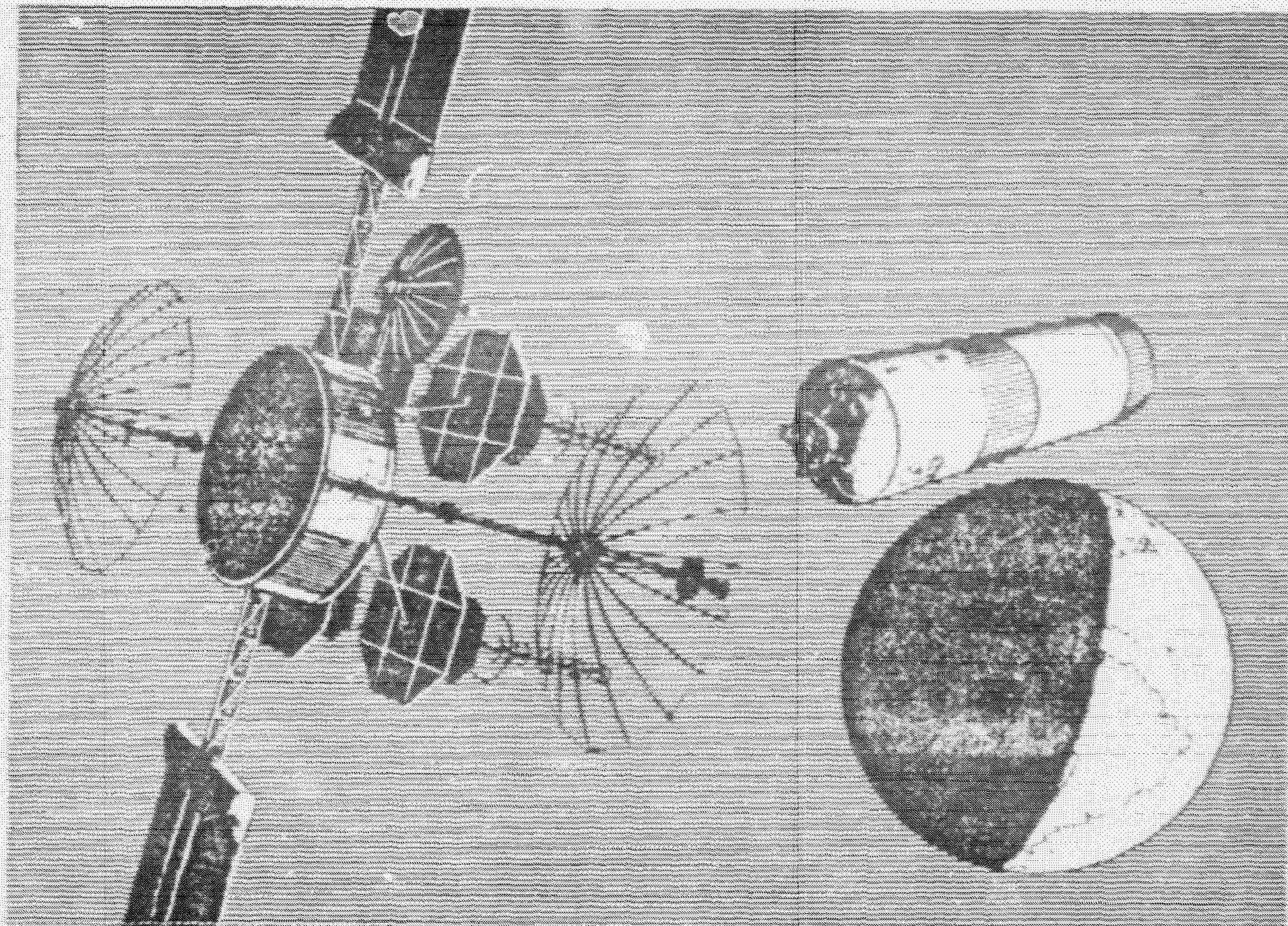


Figure 3.2-20. Geosynchronous Satellite Maintenance Sortie

Crew Rotation and Resupply—Not applicable.

Transfer and Storage—Not applicable.

Orbital Assembly, Maintenance, and Modification—Not applicable.

Consumables—A detailed estimate was not made. Consumables estimating factors used for the parametric descriptions in the appendix to this report were used to estimate consumables requirements at 350 kg (770 lb).

3.2.2.3 Transportation Requirements

3.2.2.3.1 Payload Delivery Points

The payload is delivered to a geosynchronous orbit, in some cases requiring operations in low Earth orbit, and subsequently returned to Earth either through low Earth orbit or directly.

3.2.2.3.2 Payload Delivery Options

The only option considered was delivery to geosynchronous orbit on a single OTV flight. Delivery of the system (including OTV's) employed either multiple shuttle flights or a single HLLV flight. The total payload including repair and refurbishment provisions and applicable growth allowance of 24% is 6 000 kg (13,230 lb) for the propulsive return CTV and 8 370 kg (18,450 lb) for the Apollo-mode CTV. No operational constraints were found.

3.2.2.4 Mission/Transportation Modes and Operations

3.2.2.4.1 Transportation Options

Transportation modes for the GSMS mission may be expressed in terms of disposition of the CTV upon completing the maintenance tasks. More specifically, the modes may be defined as 1) Earth orbit return and 2) direct entry.

The Earth orbit return mode has two submodes. One of these submodes uses propulsive means to return the CTV. The other uses a combination of propulsion and aerobraking.

The direct entry mode requires propulsion for deorbit and then uses the aerodynamic shape of the CTV to allow return back to a recovery area on Earth. Ballistic and winged re-entry vehicles are both candidates.

In all modes, propulsion is required to transport the CTV to geosynchronous orbit.

The resulting modes including propulsion may be summarized as follows:

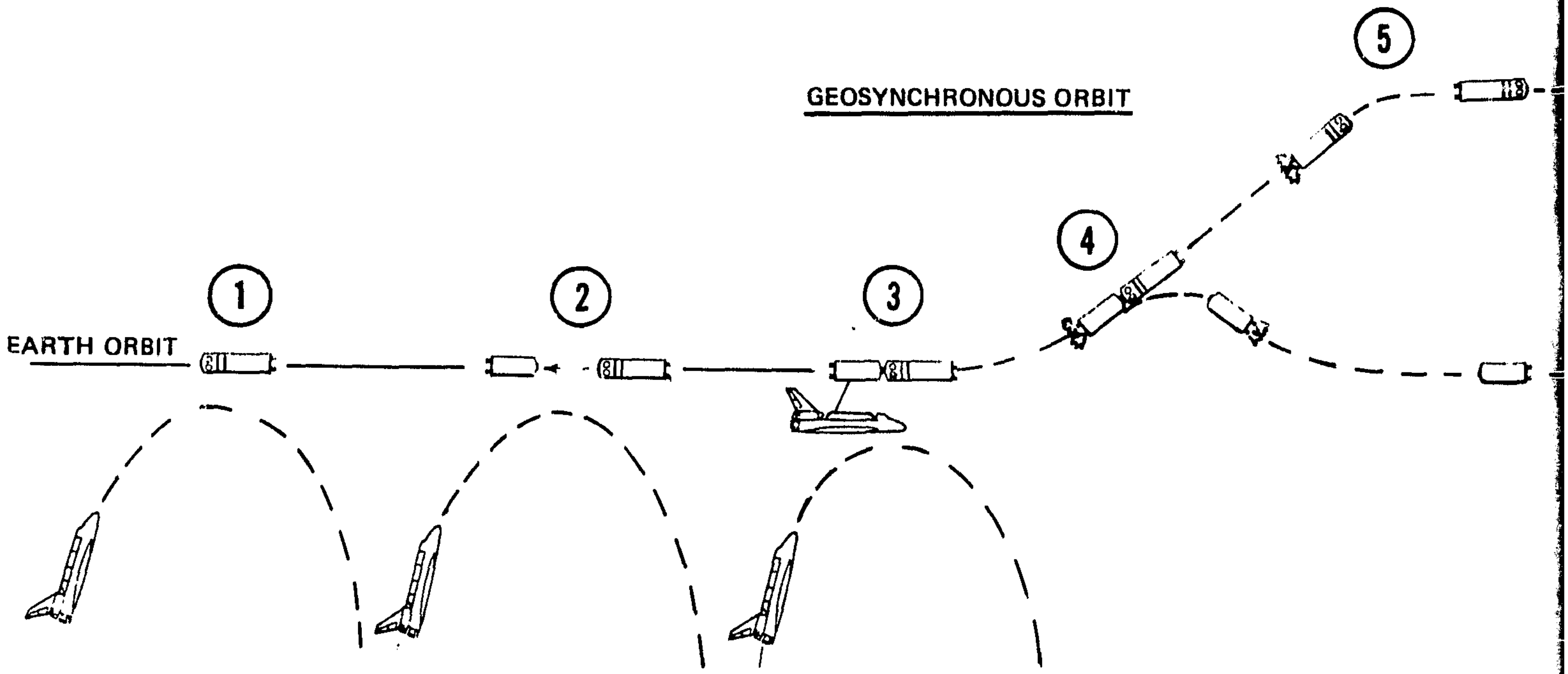
Mode	1	Earth Orbit Return
	1A	Single Stage LO_2/LH_2
	1B	Common Stage LO_2/LH_2
	1C	Common Stage LO_2/MMH
	1D	Single Stage LO_2/LH_2 with Aerobraking Device
Mode	2	Direct Entry
	2A	Single Stage LO_2/LH_2 with Aerobraking Vehicle

3.2.2.4.2 Representative Transportation Mode and System

3.2.2.4.2.1 Sequence Description

Typical sequences and operations associated with the GSMS mission are illustrated in Figure 3.2-21 using Mode 1B as a reference. The principal features of this mode are as follows:

- The CTV, two LO_2/LH_2 common stages and tanker are all delivered to Earth orbit using the space shuttle.
- Transfer to geosynchronous orbit is accomplished using two burns and requires use of both stages.
- The first stage provides approximately 75% of the first burn ΔV. After staging, the first stage returns to Earth orbit.
- The second stage completes the first burn and all of the burn required to circularize into geosynchronous orbit.
- Upon completion of the maintenance task, the second common stage is again used to deorbit and insert the CTV into a low Earth orbit where rendezvous and docking is performed with the first common stage.
- Recovery of the maintenance crew is accomplished using the space shuttle.
- Refueling of the common stages would then allow another maintenance mission to be performed.



PHASE TIME

CUM TIME

①	②	③	④	⑤
• DELIVER CTV (REPAIR CREW) AND FIRST PARTIALLY FUELED COMMON STAGE WITH SS	• DELIVER SECOND PARTIALLY FUELED COMMON STAGE WITH SS • DOCK WITH CTV/FIRST COMMON STAGE	• DELIVER ONE TANKER USING SS • COMPLETE FUELING OF BOTH COMMON STAGES	• INITIATE DELIVERY OF CTV TO GEO USING FIRST COMMON STAGE • JETTISON FIRST COMMON STAGE AND RETURN TO EARTH ORBIT	• COMPLETE DELIVERY OF CTV USING SECOND COMM STAGE • RENDEZVOUS AND DOCK WITH SATELLITE

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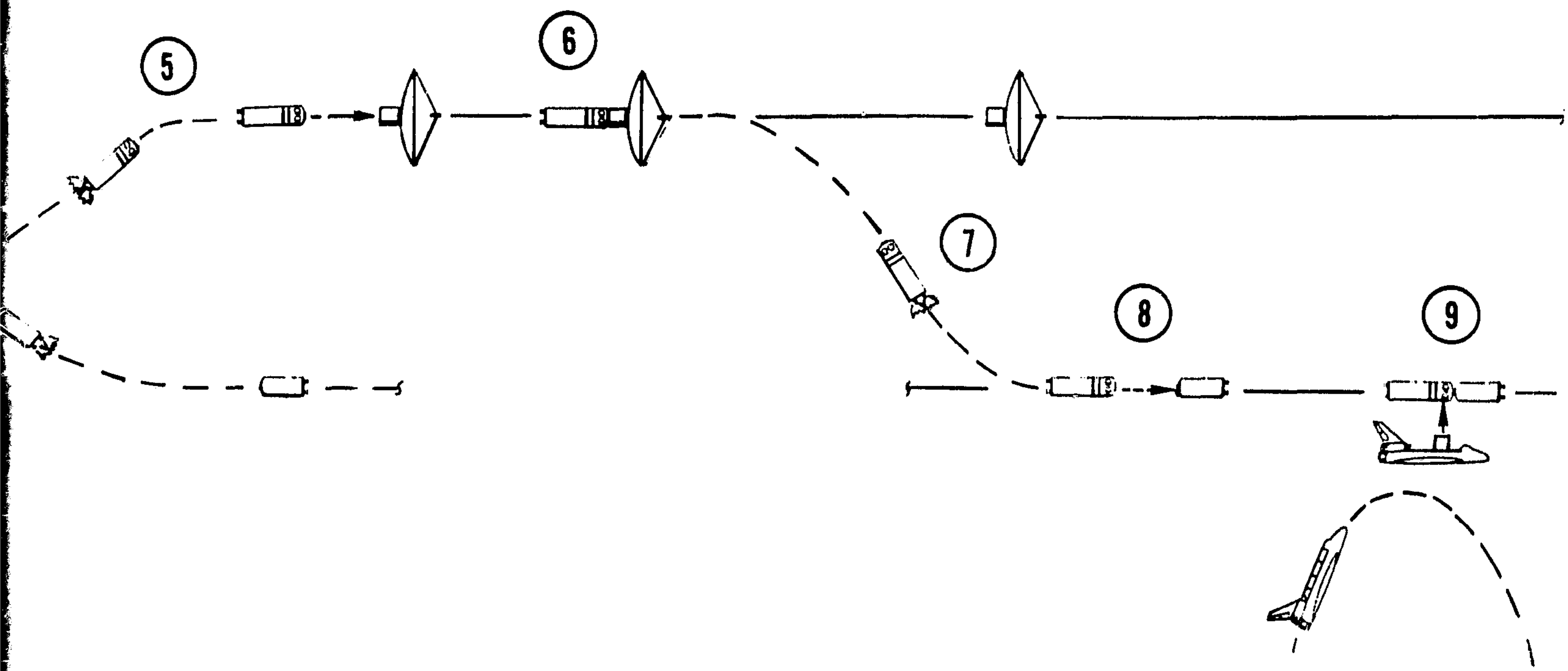


1 WEEK

	5	6	7	8	9	10
ON	• COMPLETE DELIVERY OF CTV USING SECOND COMMON STAGE • RENDEZVOUS AND DOCK WITH SATELLITE	• REPAIR AND MAINTAIN SATELLITE	• RETURN CTV TO EARTH ORBIT USING SECOND COMMON STAGE	• RENDEZVOUS AND DOCK CTV/FIRST COMMON STAGE WITH SECOND COMMON STAGE	• RETRIEVE REPAIR CREW USING SS • REPEAT	• REPEAT STEPS 3 THRU 9 FOR EACH GEOSYNC. SATELLITE REPAIR/ MAINTENANCE MISSION

FOLDOUT FRAME

Figure 3.2-21 Geosynchronous Satellite Repair Sortie

Table 3.2-14 presents key events and elapsed time, maneuver delta V and mass remaining for the representative system.

3.2.2.4.2.2 Transportation Sizing

Parametric analysis was used to size the representative system and the alternates. Figure 3.2-22 shows the results for the representative system.

3.2.2.4.2.3 Operational Factors

Sequence operations are generally similar to the GSS mission (paragraph 3.2.1.3.2.4) for propulsive modes.

3.2.2.4.2.4 Earth Launch Requirements

The representative mode requires either 3 space shuttle launches or one HLLV launch. Recovery of the crew requires a shuttle flight, or requires the last shuttle involved in launching the mission to wait in orbit for its return.

3.2.2.4.3 Transportation Options Comparison and Evaluation

3.2.2.4.3.1 Size and Performance Comparison

Propulsive Modes—Parametric transportation system performance maps with superimposed payload requirements were developed for the return to Earth orbit mode using propulsion. Results of these plots for the alternate propulsive options are shown in figure 3.2-23 and 3.2-24. Performance for the return to Earth orbit using an aerobraking kit was performed on a point design basis. Transportation analyses for the other aerobraking modes will be conducted during the Phase I extension.

Mass comparison of the transportation modes is shown in figure 3.2-25. The significantly lower mass of the return to Earth orbit using propulsion and aerobraking is the result of a 2 226 m/sec (7,302 fps) saving in ΔV. Included in this transportation mass is the penalty for the aerobraking device and for the radiation shielding.

Aerobraking Modes Discussion—Multiple pass aerobraking using a high-drag kit applied to a propulsion stage as illustrated in figure 3.2-26 allows a single stage to operate efficiently. The five days required for the 30-pass return is a severe disadvantage. Tailored aeromaneuvering stages such as the concept of figure 3.2-27 from a Lockheed study can reduce the time required.

Table 3.2-14 GSMS Mission History—LO_2/LH_2 Common Stage

Event	Cum time	ΔV		Mass remaining	
	hr	MPS	FPS	10^3 kg	10^3 lb
Initial condition (leave LEO)	0			76.4	168
Boost	0.5	2000	6561	48.2	106
Drop stage 1, skirt and stage 1 remaining fuel	0.5			42.3	93
Complete 1st burn with stage 2	0.5	547	1794	37.3	82
Ascent coast	5.5	10	32	36.8	81
Circularize	5.6	1787	5862	24.1	53
Wait-drop payload (assumed 24 hr wait in GEO)	29.6			24.1	53
Deorbit	29.7	1787	5862	15.5	34
Descent coast	34.8	10	32	15.5	34
Circularize to LEO	34.9	2447	8028	8.2	18
Stage 1 recovery				5.4	11.8
Descent-coast		10	32	5.3	11.7
Stage 1 EOI	5.5	1900	6233	3.5	7.6

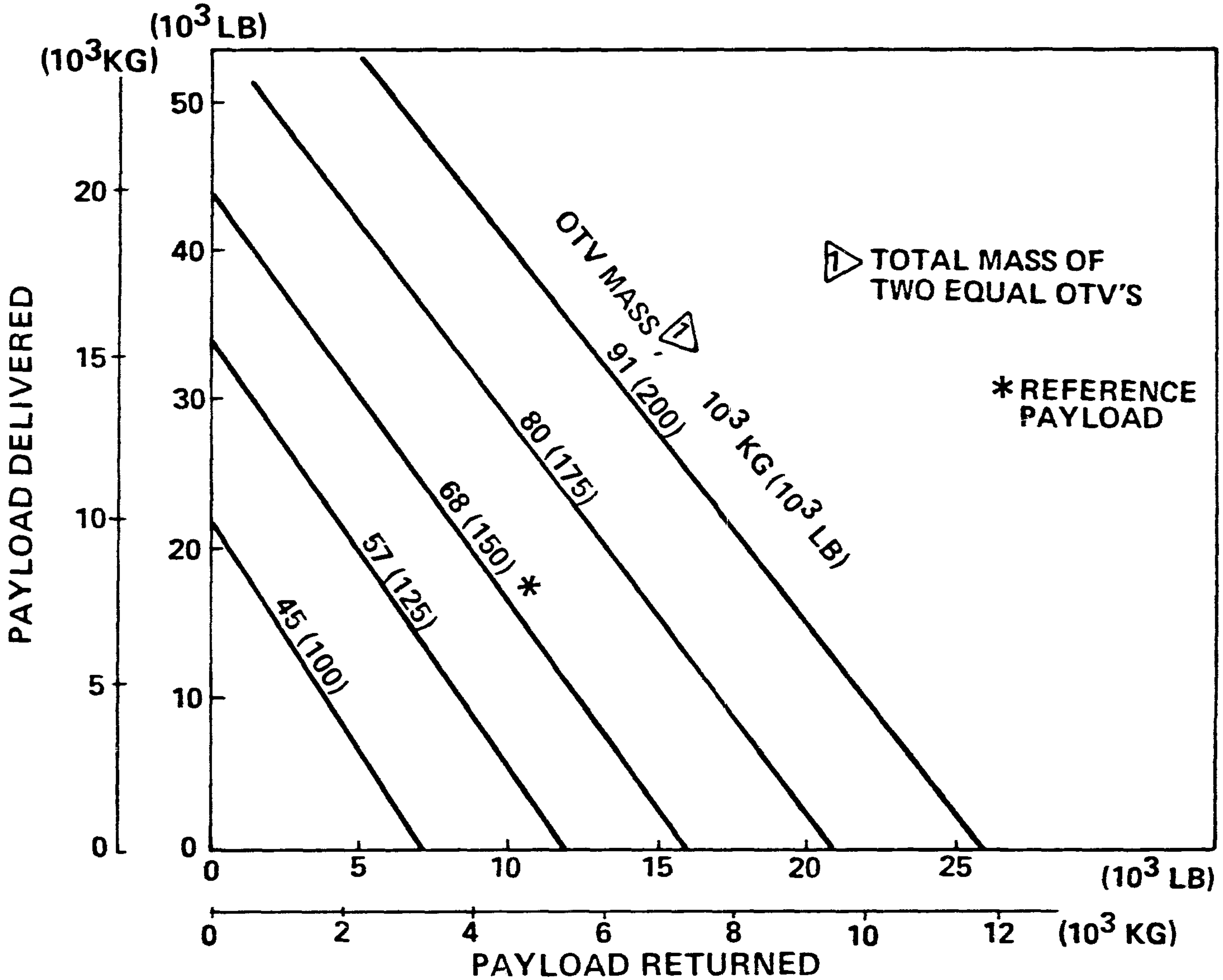


Figure 3.2-22. Common Stage LO_2/LH_2 OTV Capability for GSMS

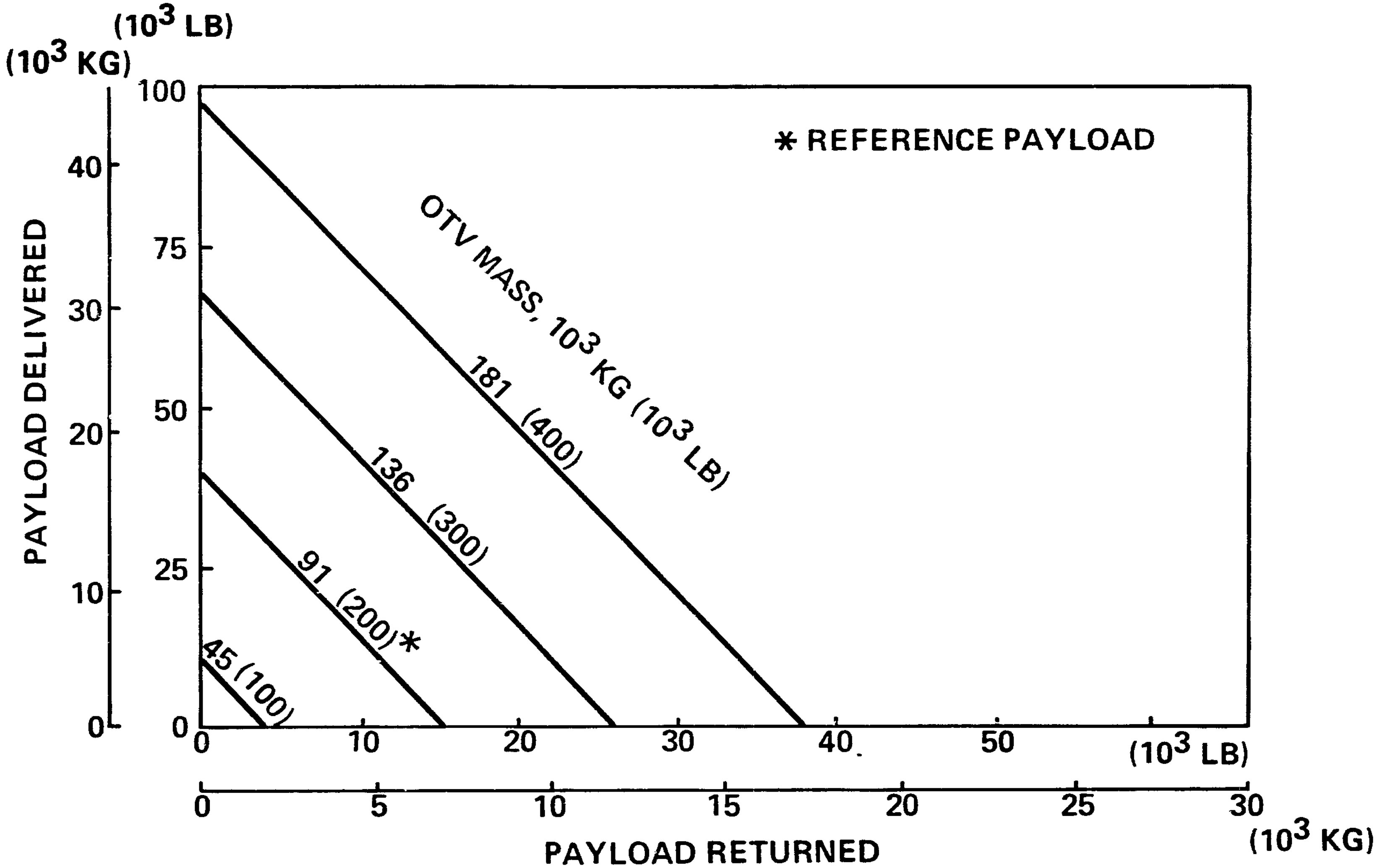


Figure 3.2-23. Single Stage LO_2/LH_2 OTV Capability for GSMS

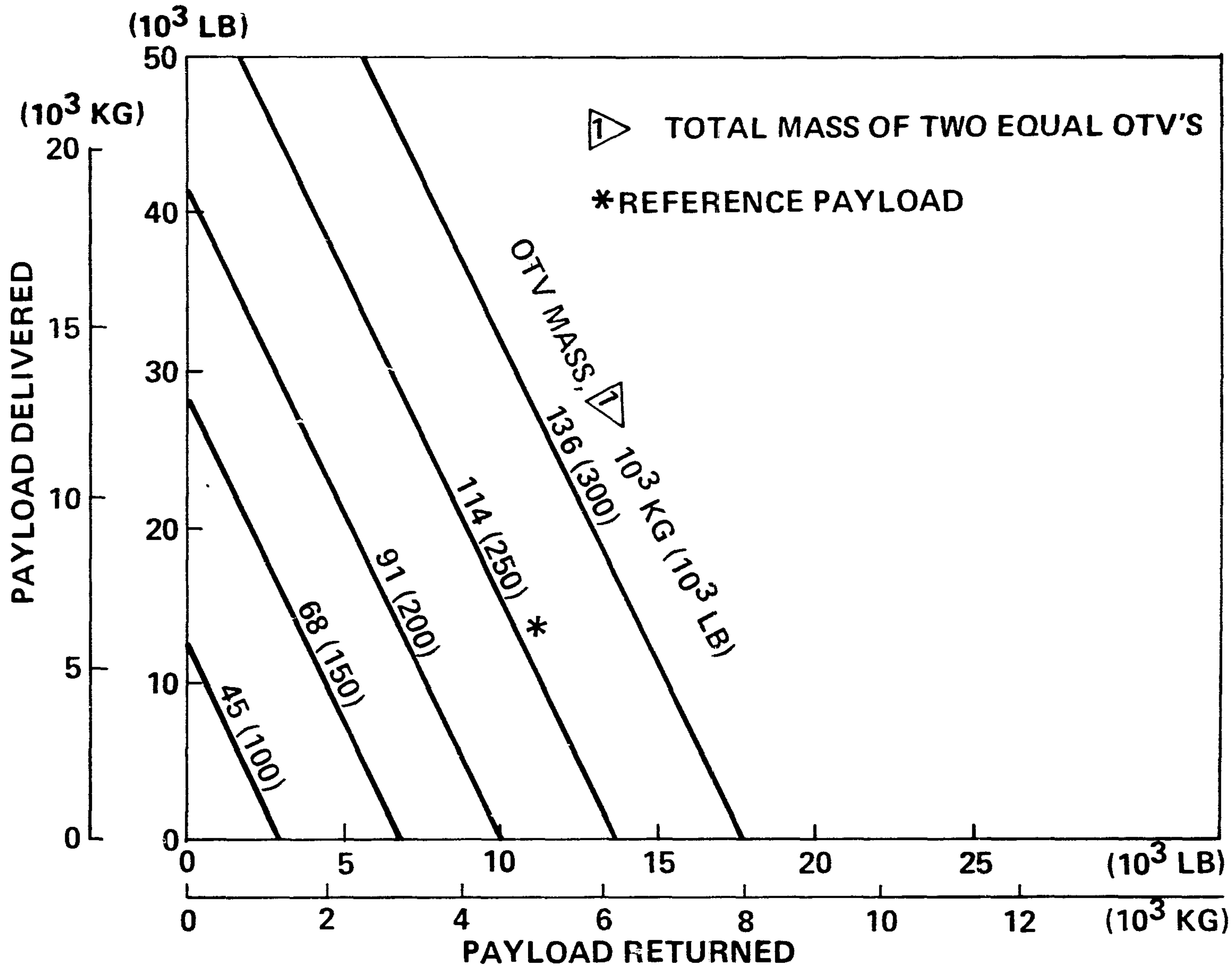
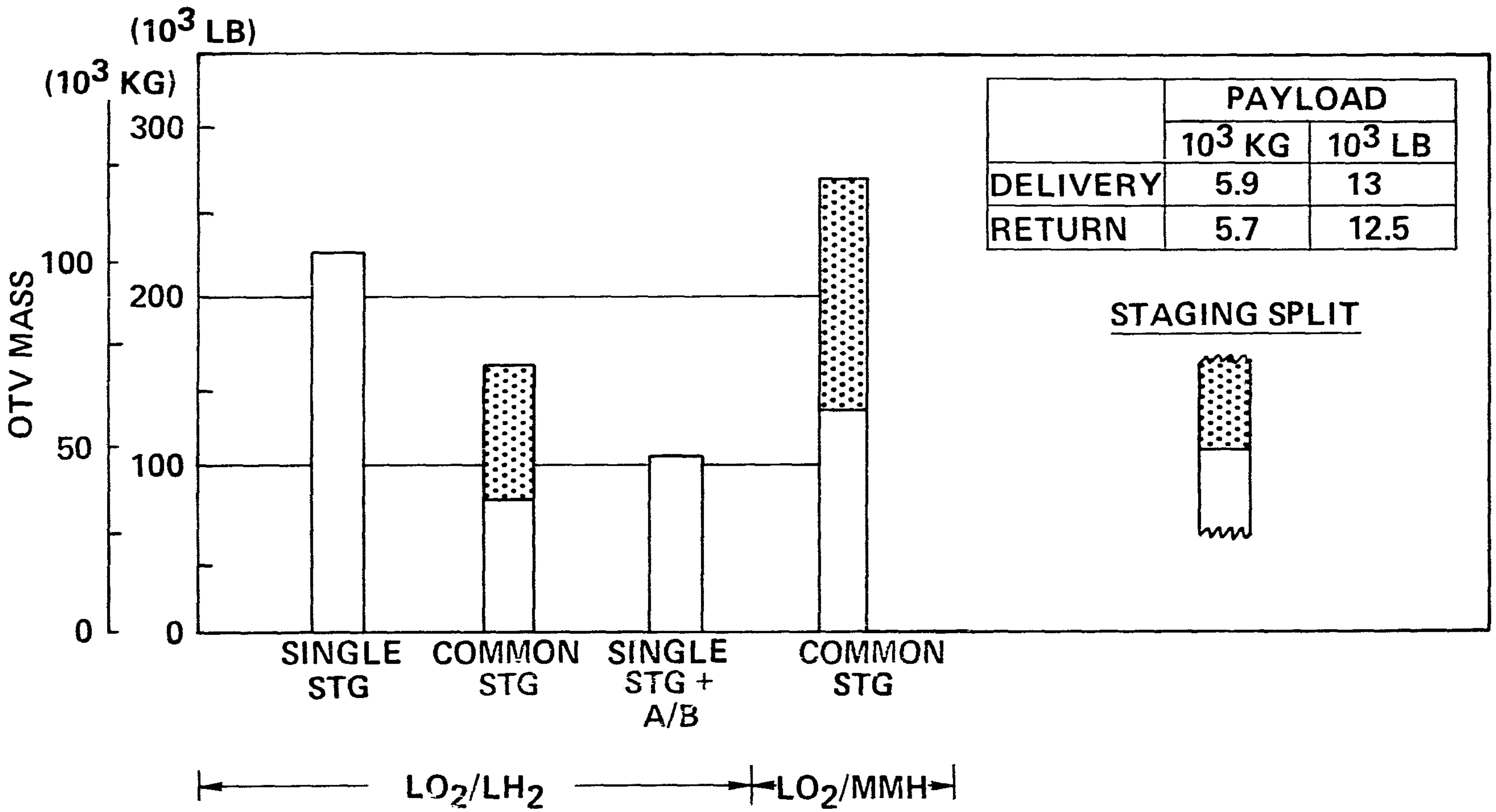


Figure 3.2-24. Common Stage LO_2/MMH OTV Capability for GSMS



	PAYLOAD	
	10^3 KG	10^3 LB
DELIVERY	5.9	13
RETURN	5.7	12.5

Figure 3.2-25. OTV Comparison for GSMS

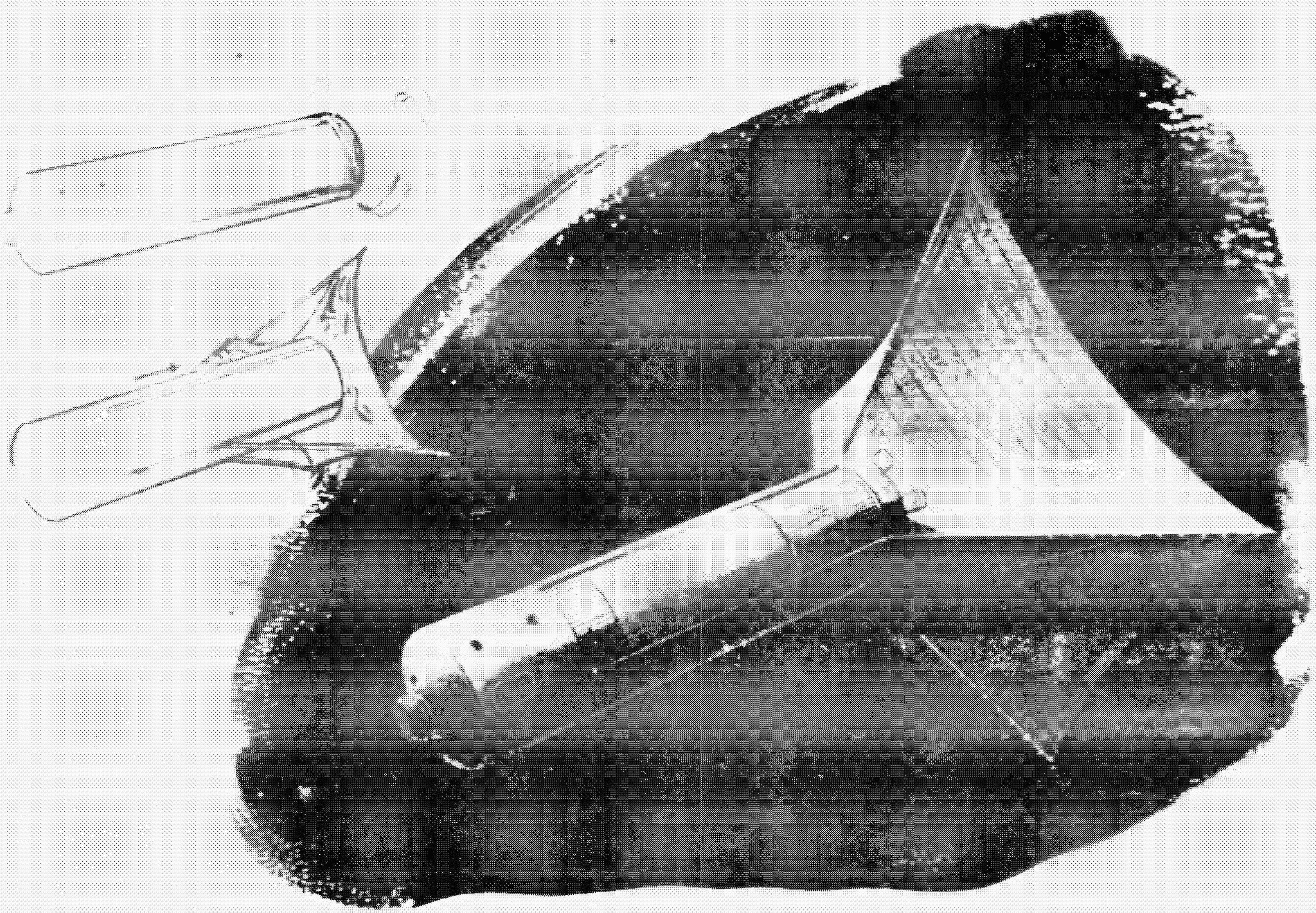


Figure 3.2-26. Aerobraking Concept

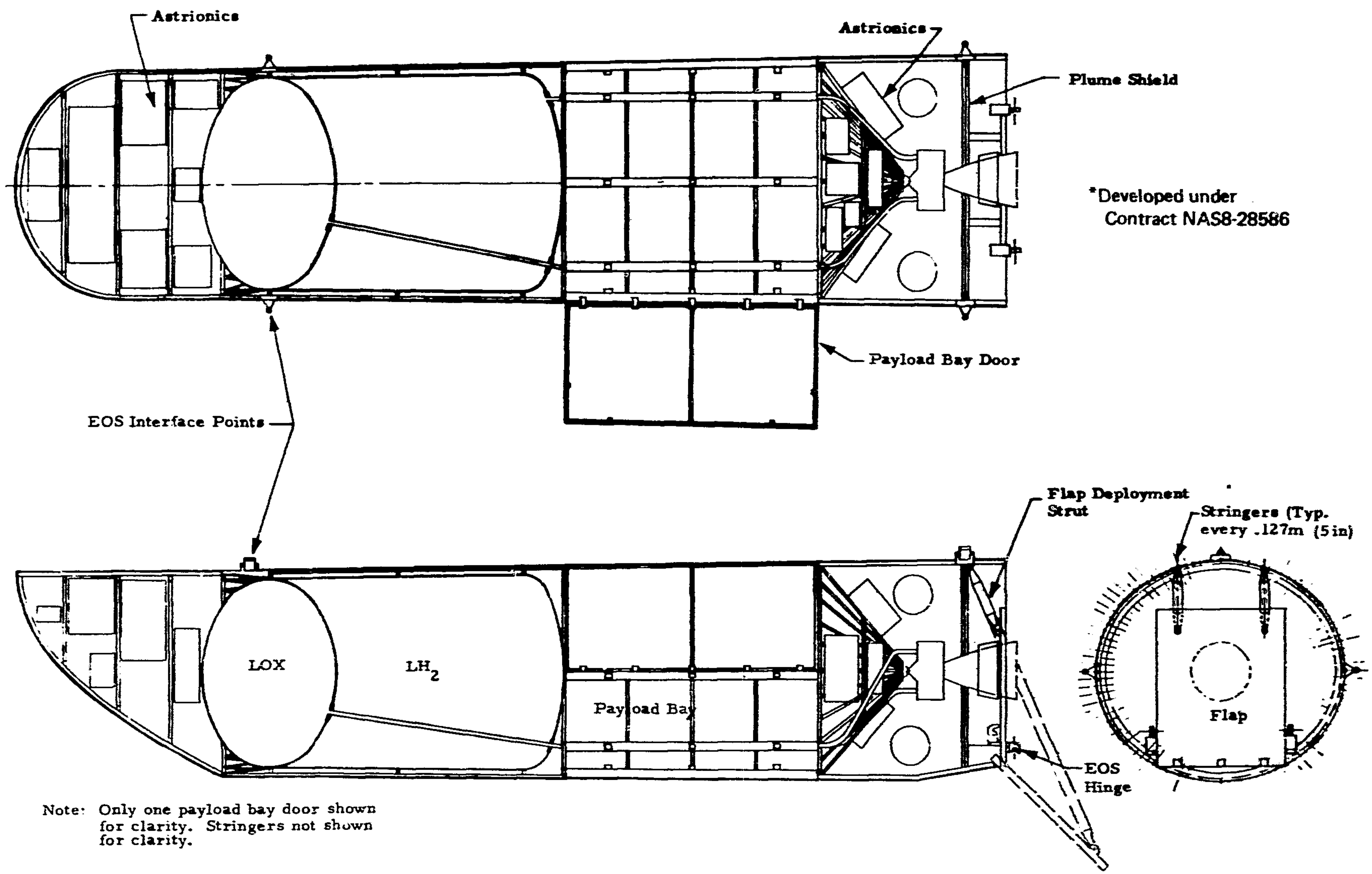


*Figure 3.2-27. Lockheed Aeromaneuvering Orbit-to-Orbit Stage (AMOOS) Concept**

Use of an Apollo-shape direct entry module allows direct return of the crew by aerobraking to sea recovery. Multiple passes are not required. The module must be propulsively deorbited from geosynchronous orbit, including a 28^{o} to 30^{o} plane change.

Direct return is possible, using an aerobraking, entry, and landing glider similar to the configuration shown in figure 3.2-28. The glider includes propulsion for a coplaner deorbit; its aerodynamic L/D provides the return plane change during aerobraking. Approximately ten aerobraking passes (2 days) are required. A mass properties analysis has not been done; mass data shown are very preliminary estimates.

3.2.2.4.3.2 Earth Launch Requirements Comparison

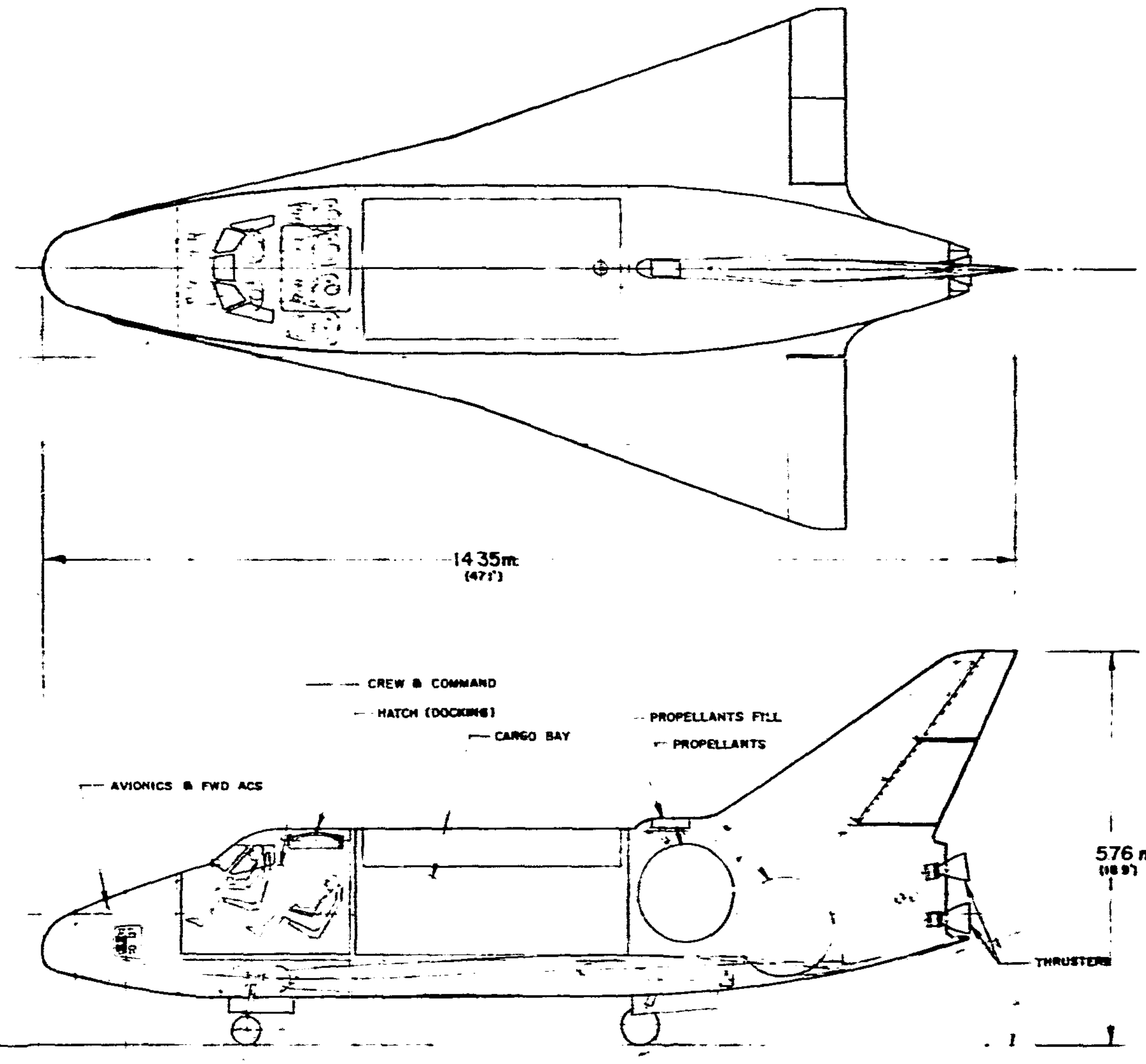
At this point in the study, no assumptions have been made relative to the number of annual flights or the time between flights. Consequently, the comparison of Earth launches is related only to that required to perform one maintenance sortie flight. This comparison is shown in figure 3.2-29 with the number of launches including those necessary for the delivery of the transportation system and CTV and maintenance provisions.

3.2.2.4.2.3 Operations Comparison

This comparison is similar to paragraph 3.2.1.4.2.3. All the geosynchronous mission/transportation modes require precision targeting for rendezvous in geosynchronous orbit and for Earth return, either to low Earth orbit or to direct or aerobraking entry. The aerobraking mode requires repeated precision targeting for each successive pass, including corrections for perturbations occurring during the aerobraking passes. Aerobraking targeting must be done quickly (one to two hours) so that corrections can be made near apogee where energy requirements are minimal. Onboard navigation and targeting is desirable for all modes and essential for aerobraking.

The return to Earth orbit mode using an aerobraking kit applied to the propulsion stage requires approximately 30 atmospheric passes in order to minimize the thermal protection penalty. Completion of the aerobraking phase will require about five days. The aerobraking mode also results in considerable radiation exposure as the vehicle passes through van Allen belts. An estimated 1 500 kg (3,300 lb) of shielding is required in order to reduce the crew radiation exposure to an acceptable level.

	WEIGHT LB	WEIGHT KG
INITIAL	41,500	18,863
PROPELLANTS	17,000	7,727
END BURN	24,500	11,136
CONSUMABLES	1,500	682
MAX LANDING WT	23,000	10,454
PERSONNEL, CARRY-ON CARGO	1,500	682
PAYLOAD BAY CARGO	3,000	1,364
RESIDUALS, RESERVES	500	227
DRY WEIGHT	18,000	8,182
PLANFORM AREA	532.3 ft²	49.54 m²
WING AREA (not includ body sec)	248 ft²	23 m²
LENGTH	47.1'	14.35 m
SPAN	25.0'	7.62 m



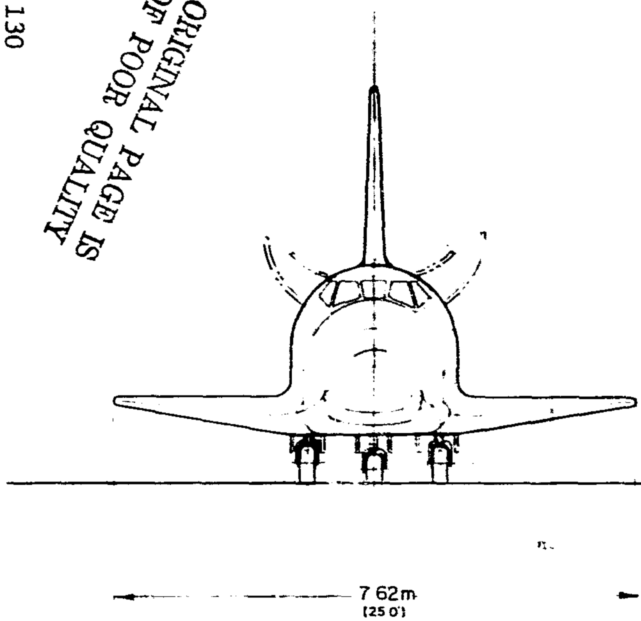


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Figure 3.2-28. Personnel/Cargo Glider

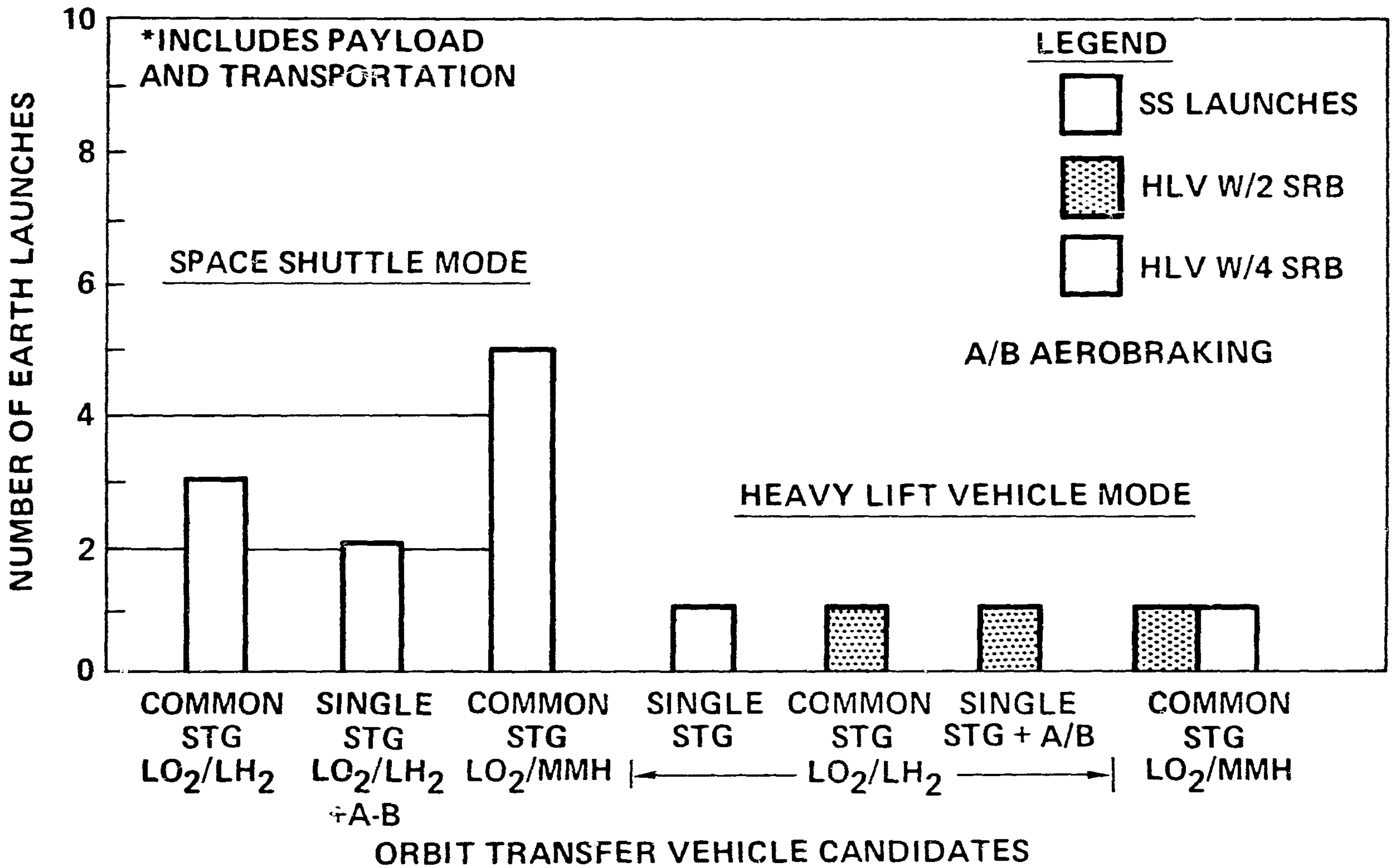

*Figure 3.2-29. Earth Launches Required for First GSMS Flight**

The glider aerobraking system requires about ten passes and two days.

Operational factors are compared in Table 3.2-15.

Table 3.2-15. GSMS Operational Factors

TRANSPORTATION MODE \ FUNCTION		NAVIGATION & TARGETING			DOCKING				
		RENDEZVOUS	LOW ORBIT REUTRN	AEROBRAKING GUIDANCE & NAVIGATION	LARGE DIAMETER INTERSTAGES	FUELING OR REFUELING	SHUTTLE RECOVERY OF HARDWARE	SEA RECOVERY	SELF RECOVERY
SHUTTLE	COMMON STAGE LO_2/LH_2	●	●		●		●		
SHUTTLE	COMMON STAGE LO_2/MMH	●	●		●		●		
HEAVY LIFT	SINGLE STAGE	●	●			●			
HEAVY LIFT	SINGLE STG AEROBRAKING	●	STAGE ONLY	●		●		APOLLO TYPE	GLIDER

3.2.2.4.2.4 Practicality Assessment

Practicality of the aerobraking kit, 30-pass mode is doubtful because of the time required and the risks, considering that a crew is involved. The other modes appear practical. Pro's and con's are compared in Table 3.2-16.

Table 3.2-16. Transportation Mode Pros and Cons

	ADVANTAGES					DISADVANTAGES		
	COMPATIBLE WITH SHUTTLE LAUNCH AND RECOVERY	NO STAGE-TO-STAGE DOCKING	FULLY REUSABLE	ONLY ONE STAGE TO DEVELOP	NO LH_2	COMPLEX, POTENTIALLY RISKY MISSION PROFILE	TAILORED DEVELOPMENT WITH LIMITED COMMONALITY WITH OTHER MISSIONS	TWO ACTIVE VEHICLES TO TRACK
LO_2/LH_2 SINGLE STAGE		X	X	X				
LO_2/LH_2 COMMON STAGE	X		X	X				X
LO_2/MMH COMMON STAGE	X		X	X	X			X
AEROBRAKING KIT	X	X	?	X		X		
AEROMANEUVERING STAGE	X		X	X		X	X	
APOLLO MODULE	X	X		X				
AEROBRAKING GLIDER			X	X				

3.2.3 OPTIONAL GEOSYNCHRONOUS PAYLOAD STUDIES

The concepts described below as option payloads were developed as a part of this study. The mass communicatións satellite was examined by a Boeing in-house study about 7 years ago and updated for this study. The LST occulting disk has been discussed in the literature since about 1962, but no substantive analyses were found.

These option payloads represent concepts in the formative stage. As such, they are representative of advanced payloads that might be operational during the time frame of manned geosynchronous operations. They do not, however, constitute selections or recommendations as to advanced payloads. Both concepts need further study to develop increased confidence in characteristics and capabilities.

The LST occulting disk mission requires placement of an LST in a very high Earth orbit. Lunar trojan points are representative. Although this orbit is not geosynchronous, the transportation requirements are more nearly equivalent to geosynchronous missions than to the others discussed in this report. The occulting disk mission is therefore included in this section.

3.2.3.1 Mass Communications Satellite

The mass communications satellite concept proposes to provide a system of large sophisticated satellites as "switchboards in the sky" serving subscribers in a manner analogous to the hard-wired and microwave telephone links of today. It is possible that users of this system could communicate through hand-held two-way units to other users anywhere in the world. Whether this could be achieved is doubtful, but telephone units capable of some antenna directivity could communicate effectively through such satellites. A concept of a mass communications satellite is shown in figure 3.2-30.

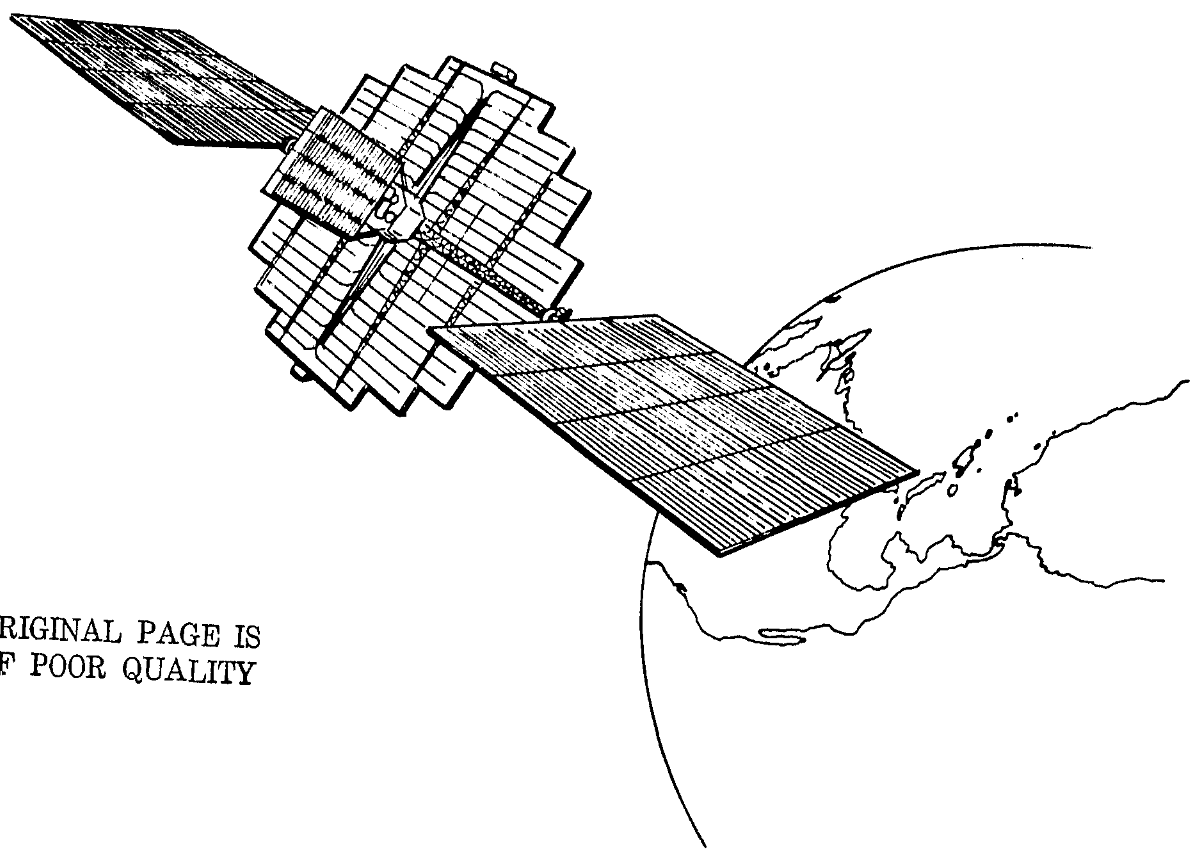

Figure 3.2-30 Mass Communications Satellite Concept

These satellites, although unmanned, would be large and complex, requiring periodic man-directed service on orbit to achieve an adequate level of system availability. From 1 to 40 satellites would comprise the system, depending on the level of subscription. Each satellite is expected to weigh about 40 000 kg (88,000 lb). It is estimated that manned service missions on the order of twice a year would be desirable. Each service mission (see paragraph 3.2.2) would visit all satellites needing service.

3.2.3.2 LST and Occulting Disks

The LST occulting disk mission is postulated as a means to achieve direct observation of planets around nearby starts. The LST unaided might be able to observe large planets around a few (one to five) of the brightest neartest stars. Such planet observations will be extremely difficult because the parent star will be brighter by a factor of 10^8 to 10^{10} (20 to 25 magnitudes) than the planet. Diffracted light from the steller image will be more intense by a factor of 10^2 to 10^6 than the planet image sought.

In 1962, Spitzer in a paper on extrasolar planetary systems–discussed the idea of placing an occulting disk at a great distance from a space telescope to blot out the bright star image while allowing the planet image to be seen (figure 3.2-31). The improvement in detectability by this technique was estimated in a concept study for the FSTSA study and found sufficiently promising that the LST/occulting disk was incorporated as a representative advanced payload for geosynchronous operations.

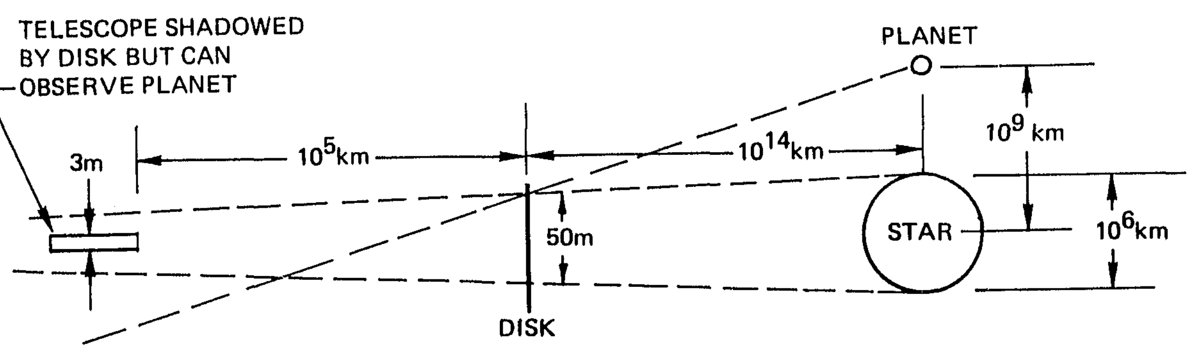

Figure 3.2-31: Concept of the Occulting Disk (Not to Scale) - - - Distances Shown are Typical

Each time an observation is attempted, the occulting-disk spacecraft must fly to the required occultation position (different for each target star), remain in this position under active control of the LST for roughly 1 hour, and then return to the vicinity of the LST to await the next attempt. The disk must be a great distance—typically 10^5 km (54,000 nmi) from the telescope. Unless this distance is small compared to the distance of the telescope from the Earth, the propulsion requirements for maneuvering the disk are enormous. Therefore, location of an LST at a lunar trojan libration point 380 000 km (250,000 nmi) from Earth was selected as representative.

The disk spacecraft, shown conceptually in figure 3.2-32, will have a gross mass of about 3 000 kg (6,600 lb) including 1 000 kg (2,200 lb) of propellant (sufficient for one observation maneuver). It would normally be attached to a propellant storage facility co-orbital with the LST. The number of potential target stars is roughly 30: an average of 5 observation attempts for each is a representative figure. Assuming the observation program to extend over a 5-year period, the following transportation requirements are estimated.

Initial—The initial transportation requirements include—

- Placement of a 15 000 kg (33,000 lb) LST at the libration point
- Placement of the occulting-disk spacecraft and support facility (20 000 kg [44,000 lb] total) at the same libration point

Recurring—Manned service mission twice a year to the libration point in a mission similar to the GSMS mission (paragraph 3.2.2), but also including 10 000 to 15 000 kg (22,000 to 33,000 lb) of propellant for the occulting-disk spacecraft.

Final—Return of the LST to low Earth orbit to Earth for continued use in other science program.

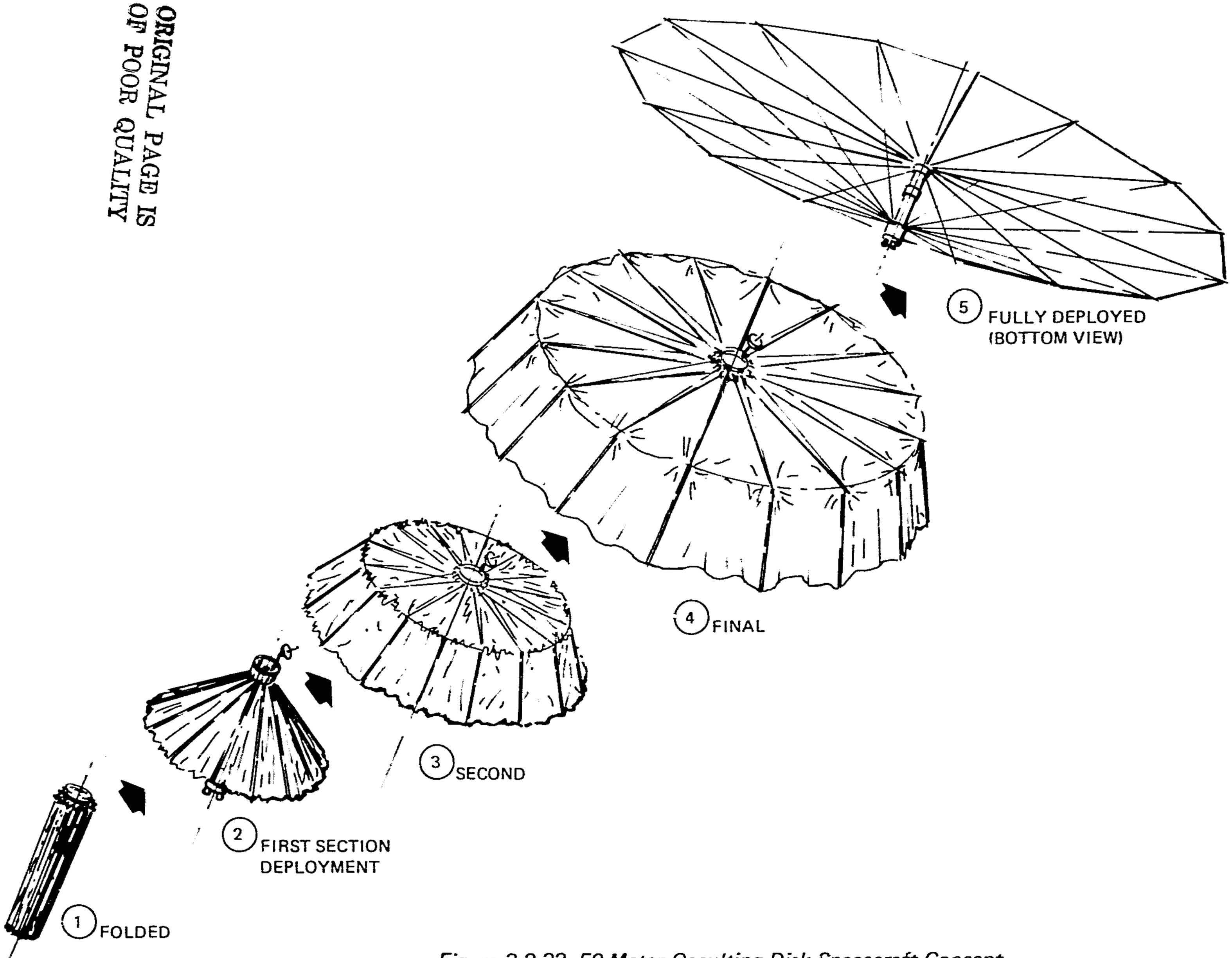

Figure 3.2-32. 50-Meter Occulting Disk Spacecraft Concept

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3.3 INDEPENDENT LUNAR SURFACE SORTIE PROGRAM

This program is represented by a single independent lunar sortie mission concept with several transportation approaches.

3.3.1 INDEPENDENT LUNAR SORTIE MISSION

3.3.1.1 Mission Summary

3.3.1.1.1 General Description

The term "independent" signifies that each mission is self supporting as were the Apollo lunar missions. Logistics flights, support missions, or systems are not required. A shuttle flight may be required to return the mission crew to Earth after their return to Earth orbit, depending on the transportation mode selected.

There is no principal reference for this mission. Mission equipment data are based primarily on the OLS study and the Mimosa study.

Each mission leaves Earth orbit, transfers to the moon, enters a lunar orbit, lands four men and 4 500 kg (10,000 lb) of mission equipment on the lunar surface for a 14-day exploration stay, and then returns to Earth. A representative independent lunar sortie configuration is illustrated in Figure 3.3-1.

3.3.1.1.2 Mission Assumptions and Constraints

Nominal mission assumptions and constraints are summarized in table 3.3-1. The polar orbit is considered representative as it phases well with a 14-day surface stay. Alternative orbits are not excluded. In particular, a halo parking orbit will provide continuous communication via relay for surface missions on the back side of the moon (i.e., not visible from Earth). The constraints are intentionally kept general in order to not exclude any of the several transportation modes potentially applicable to this mission.

3.3.1.2 Mission System Descriptions

3.3.1.2.1 Mission Options

The selection between return to Earth orbit and direct return to Earth determines the nature of payloads for the independent lunar surface mission. The options are shown in figure 3.3-2. The return to Earth orbit mode was selected as representative for transportation analyses. Comparison data were developed for the direct mode.

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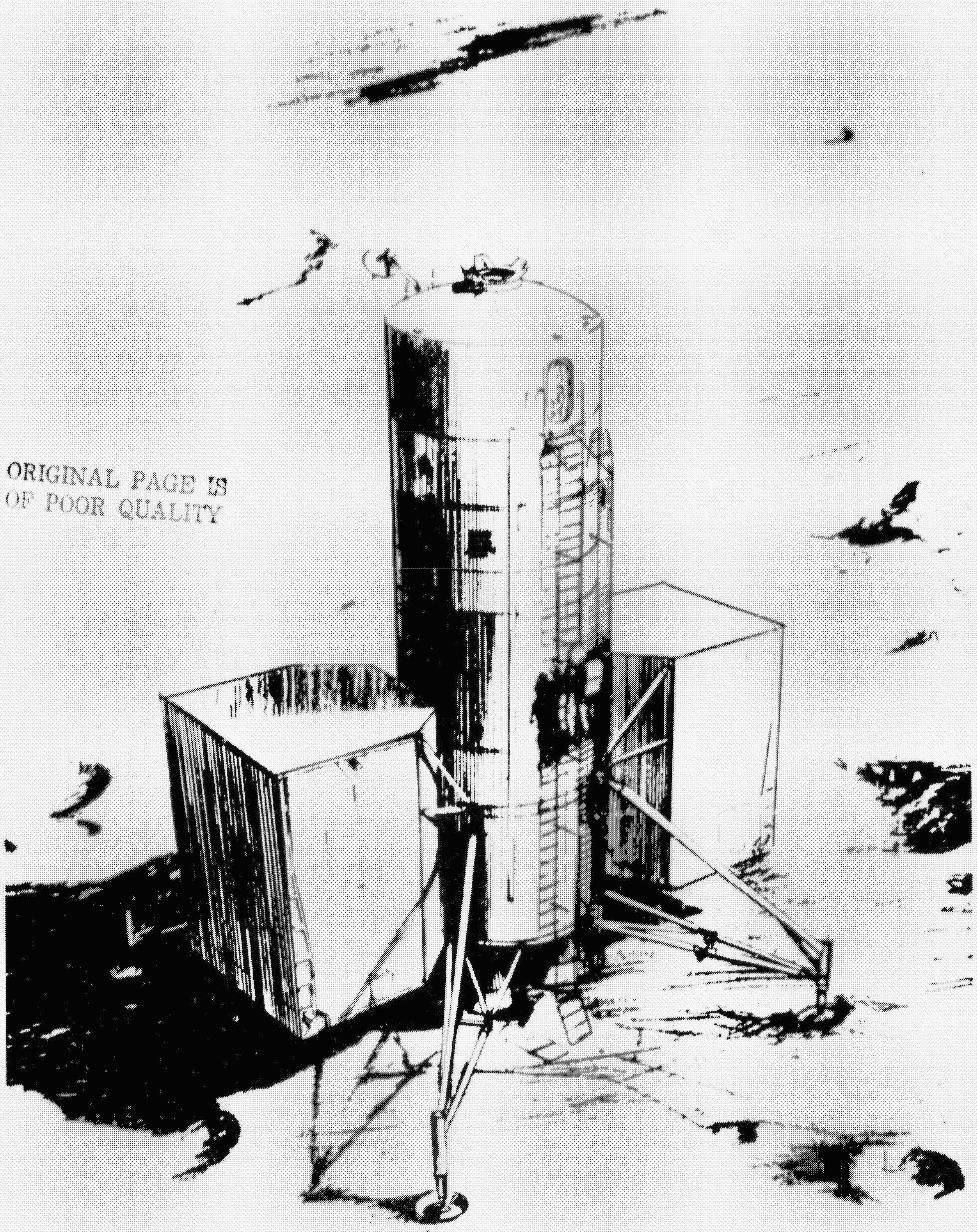


Figure 3.3-1: Independent Lunar Surface Sortie Concept

Table 3.3.1 Assumptions and Constraints

MISSION	OBJECTIVES	MISSION ASSUMPTIONS & CONSTRAINTS
• MANNED LUNAR EXPLORATION (WITHOUT SUPPORTING LUNAR ORBIT STATION)	• IN-DEPTH EXPLORATION OF SELECTED LUNAR AREAS	• FOUR MAN CREW CAPACITY (MISSION MODULE) • MISSION MODULE, ASSOCIATED WITH PROPULSION STAGE (S) FOR DESCENT AND ASCENT • 14 DAY STAY TIME • TWO MAN ROVER WITH 10 DAYS LIFE SUPPORT CAPABILITY 100KM OUT AND RETURN RANGE CAPABILITY • EXPERIMENT STATION WITH ALSEP TYPE CAPABILITIES • BACK SIDE CAPABILITY WITH COMMUNICATION THROUGH RELAY SATELLITE • SAMPLE RETURN CAPACITY OF 500 KG • 30M DRILL CAPACITY • TOTAL LUNAR SURFACE ACCESSIBLE EXCEPT FOR TBD

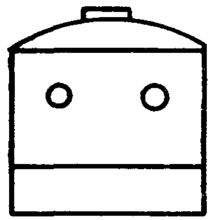
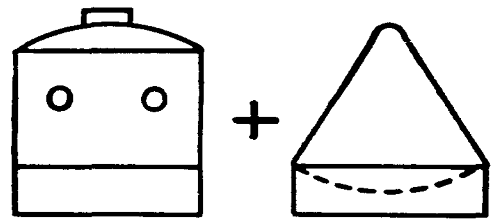
MODE OPTIONS	RETURN TO EARTH ORBIT	DIRECT RETURN TO EARTH (APOLLO TYPE LOR)
CREW QUARTERS	 • CEM • 14-DAY SURF CAPAB • 41-DAY DESIGN CAPAB • 7920 KG (17,470 LB)	 • CEM • 14 DAY SURF CAPAB • 21 TOTAL CAPAB • 6695 KG (14,765 LB) + • EARTH RETURN MODULE • 7 DAY CAPAB • 6870 KG (15,145 LB)
EXPLORATION AND SCIENCE	• MANNED ROVER, FLYER AND EXPERIMENTS • 4535 KG (10,000 LB)	
LTV OPTIONS	• SINGLE STAGE – LO2/LH2 • SINGLE STAGE – LO2/MMH	
OTV OPTIONS (TLI, LOI, TEI, EOI)	• SINGLE STAGE – LO2/LH2 • COMMON STAGE – LO2/LH2 • COMMON STAGE – LO2/MMH • 2-1/2 STAGE – LO2/MMH	• TWO STAGE – LO2/MMH

Figure 3.3-2. ILSS Mission Modes and Requirements

3.3.1.2.2 Payload Descriptions

3.3.1.2.2.1 Crew and Equipment Module

The crew and equipment module (CEM) supports the crew for the entire mission in the return-to-Earth orbit (REO) mode and for all of the mission except Earth return entry and landing for the Apollo mode. For the ILS mission representative transportation mode, the CEM is capable of docking on the forward end of a LSV and equipment module while attached by a separation joint to a second LSV/EM; the CEM draws its power and consumables from either or both equipment modules. the CEM is shown in Figure 3.3-3. A mass summary for 41-day total capability is stated in table 3.3-2.

Table 3.3-2. 41-Day ILSS CEM Mass Summary

Item	kg	lb
CEM inerts	4860	10,710
Crew, gear and reserves	1160	2,560
Consumables	1900	4,200
Total	7920	17,470

This represents the maximum mission duration; a more typical duration is 30 days.

For the Apollo mode the CEM is designed for a nominal 34-day total capability, resulting in the mass data summarized in table 3.3-3.

Table 3.3-3. 34-Day ILSS CEM Mass Summary

Item	kg	lb
CEM inerts	4535	10,000
Crew, gear and reserves	1160	2,560
Consumables	1575	3,470
Total	6695	16,030

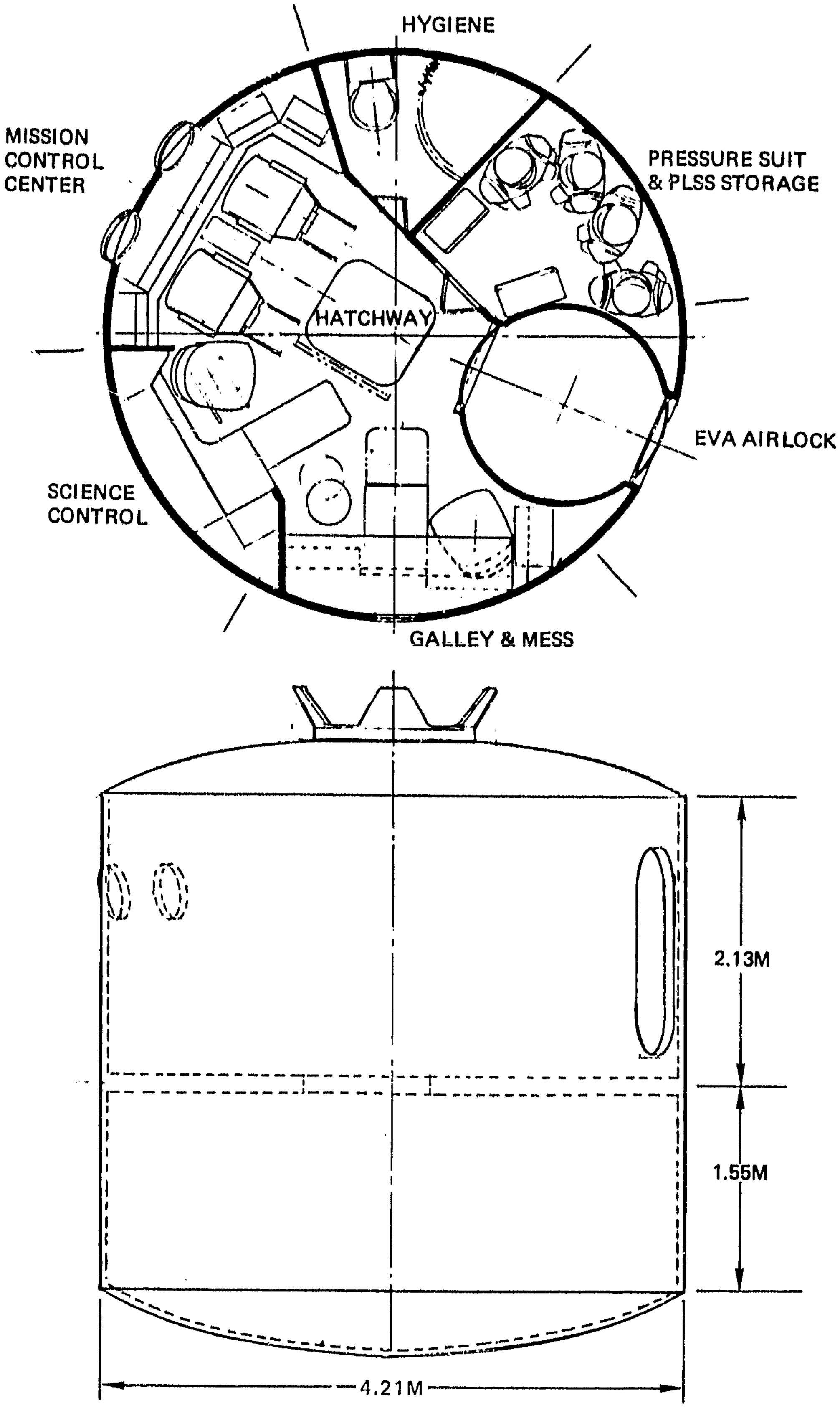


Figure 3.3-3. Crew Transport Module

3.3.1.2.2.2 Earth Entry Module

The Apollo/LOR mode Earth-entry module carries the crew to the Earth's surface from a trans-Earth trajectory by aerodynamic braking, followed by parachute descent. It could be used by the crew as a habitat during the translunar, lunar orbit, and trans-Earth mission phases. However, it is assumed in this study that the CTM is used for these purposes, allowing the EEM to be more compact. As an option, a fifth man may be added to the mission as an EEM solo pilot during the lunar surface phase to provide a backup to LSV-active rendezvous and docking after lunar ascent.

Also considered was a direct mode, landing the Apollo-type EEM directly on the moon along with return propulsion. In this approach, no crew equipment module is used so the Earth entry module houses the crew for the total mission. It should be noted that the habitable volume of the EEM is about 1/3 that of the CEM due to the diameter restriction of the shuttle. This mission mode is therefore not fully responsive to mission requirements and is shown only for comparison.

Masses of the EEM are stated in table 3.3-4.

Table 3.3-4. Earth Entry Module Masses

	APOLLO MODE		DIRECT MODE	
	KG	LB	KG	LB
EEM DRY WEIGHT	5 535	12,200	5 535	12,200
SERVICE MODULE EQUIPMENT (DRY)*	1 010	2,230	1 010	2,230
CREW, GEAR & RESERVES	1 160	2,560	1 160	2,560
CONSUMABLES	325	715	1 900	4,200
TOTAL	8 030	17,705	9 605	21,190
NET PAYLOAD WITH CREW ACCOUNTED IN CEM	6 870	15,145	—	—

*CARRIED IN PROPULSIVE STAGE; NOT PART OF ENTRY PAYLOAD

3.3.1.2.2.3 Surface Exploration Payload

For the 14-day surface mission, 4 535 kg (10,000 lb) of exploration equipment are provided. Any combination of equipment can be accommodated. A representative payload is summarized in table 3.3-5.

Table 3.3-5. Surface Payloads

Payload	kg	lb
Lunar Roving Vehicle	1 900	4,190
Transport and deployment pallet	500	1,100
Surface experiments	1 600	3,530
Experiments canister	535	1,180
Total	4 535	10,000

3.3.1.2.2.4 Crew Rotation and Resupply Payloads

Not applicable.

3.3.1.2.2.5 Consumables

Typical consumables estimating factors are stated in table 3.3-6.

Values do not include reserves. Water is assumed derived from fuel cells.

Table 3.3-6. ILSS Consumables

Item	kg per manday	lb per manday
Breathing O_2	1.2	2.65
Food	0.9	2.0
Cabin leakage	1.5	3.3
Power (fuel cells)	5.5	12.0
Lioh and miscellaneous	2.5	5.5
Total	11.6	25.45

3.3.1.2.2.6 Mass Summary

The mass summary for the ILSS modes investigated is presented in table 3.3-7.

3.3.1.2.2.7 Pickup Points and Transportation Constraints

Pickup Points—The CEM and EEM are designed to be launched fitted to propulsion stages. They also incorporate docking fixtures capable of providing sufficient structural strength for orbit transfers. Surface payloads are assumed to be handled as depicted in figure 3.3-4.

Table 3.3-7. ILSS Mass Summary

Event	REO Mode		Apollo Mode		Direct Mode*	
	10^3 kg	10^3 lb	10^3 kg	10^3 lb	10^3 kg	10^3 kg
TLI						
• CEM	4.86	10.71	4.54	10.0	—	—
• EEM	—	—	5.54	12.2	5.54	12.2
• SM	—	—	1.01	2.23	1.01	2.23
• Surface payloads	4.54	10.0	4.54	10.0	4.54	10.0
• Crew, equipment, reserves	1.16	2.56	1.16	2.56	1.16	2.56
• Consumables	1.9	4.2	1.9	4.2	1.02	2.25
Total	12.46	27.47	18.68	41.19	13.26	29.23
Lunar Descent						
TLI (less)						
• Transfer consumables	(.47)	(1.03)	(.47)	(1.03)	(.47)	(1.03)
• Vehicle in L.O.	—	—	(6.55)	(14.43)	—	—
Total	11.99	26.44	11.67	25.72	12.80	28.21
Lunar Ascent						
Lunar descent (less)						
• Surface consumables	(.65)	(1.44)	(.65)	(1.44)	(.65)	(1.44)
• Surface payloads	(4.54)	(10.0)	(4.54)	(10.0)	(4.54)	(10.0)
• Plus ret. science	.20	.44	.20	.44	.20	.44
Total	7.00	15.45	6.68	14.73	7.81	17.22
TEI						
Lunar Ascent (less)						
• Consumables	(.47)	(1.03)	(.47)	(1.03)	—	—
• Vehicle in L.O.	—	—	(4.54)	(10.0)	—	—
• Plus vehicle for return	—	—	6.55	14.43	—	—
Total	6.54	14.42	8.23	18.14	7.81	17.22
EOI or Entry						
TEI less consumables	(.47)	(1.03)	(.47)	(1.03)	(.47)	(1.03)
Total	6.07	13.39	7.76	17.11	7.34	16.19

Note:
TLI — Translunar injection
TEI — Trans-Earth injection
EOI — Earth orbit capture

* Does not provide required volume for crew and equipment

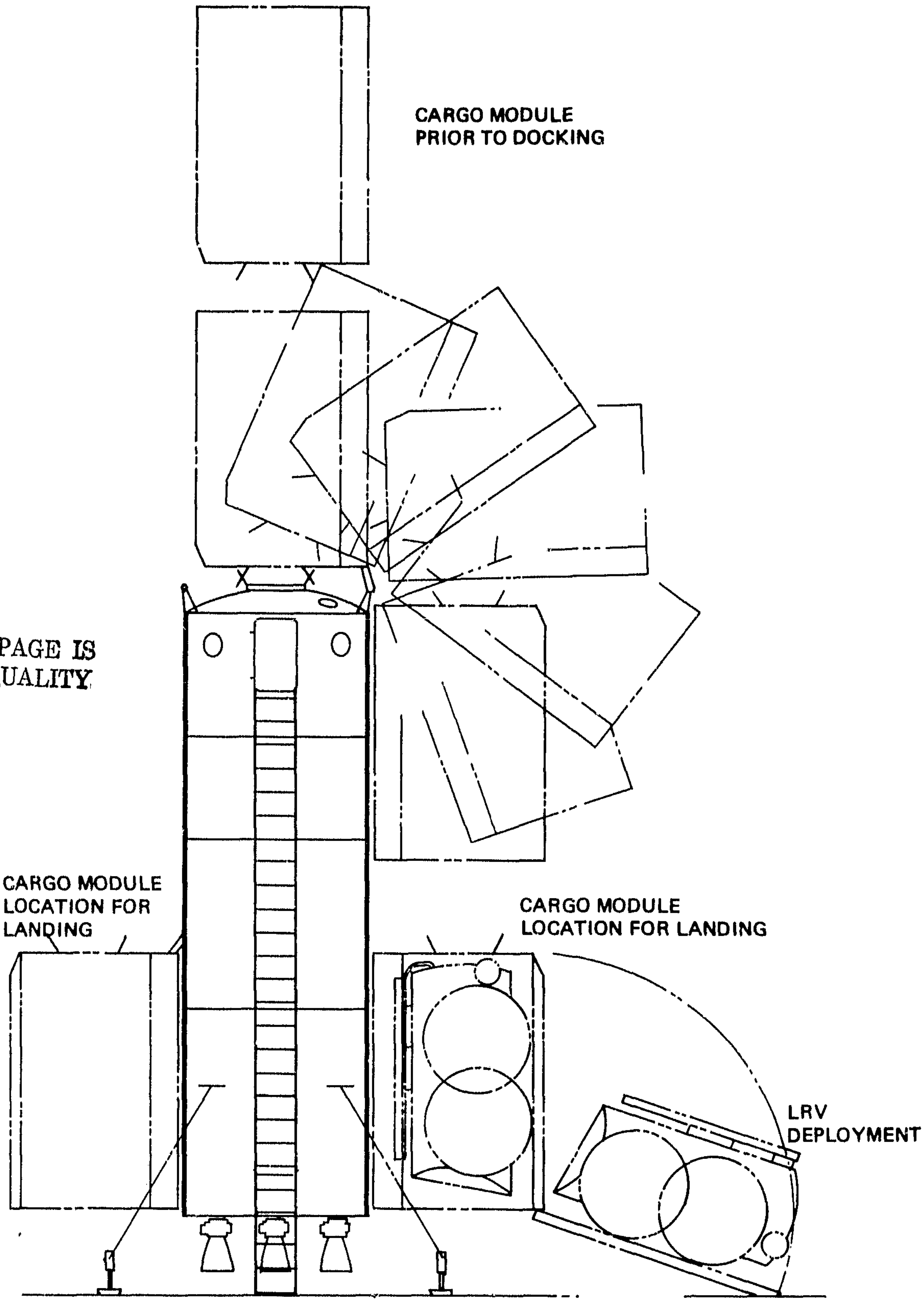


Figure 3.3-4: Cargo Deployment from Lunar Sortie Vehicle

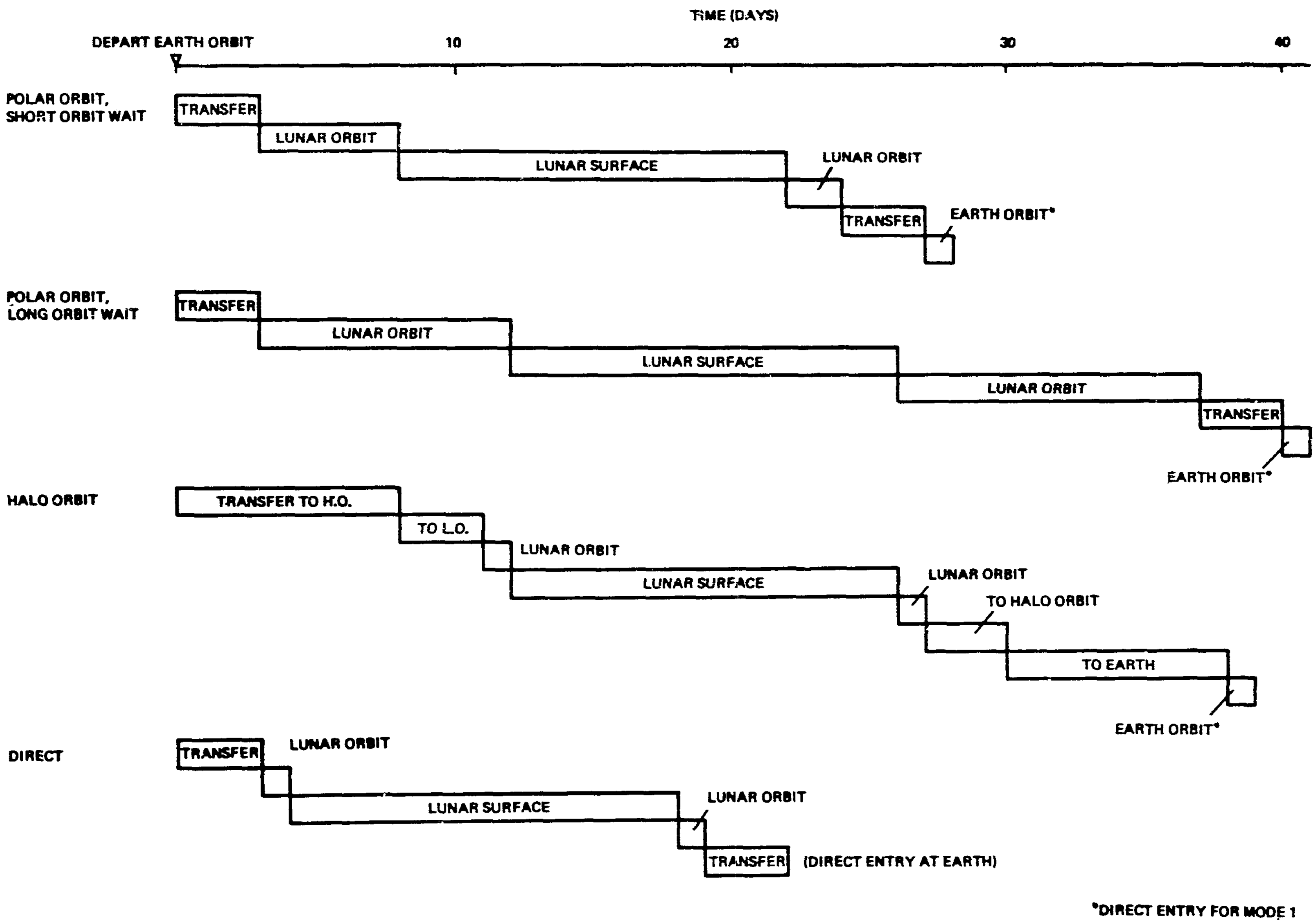


Figure 3.3-5: Comparison of Timelines for Alternative Independent Lunar Surface Mission Modes

Polar Orbits—Orientation of the polar lunar orbit results in a nominal stay time in lunar orbit of 21 days, including the 14-day surface stay period. Some surface site longitudes will require a lunar orbit time of about 34 days. Total mission time, including 3-day transfers to and from the Moon, is approximately 28 days or 41 days for the 34-day orbit stay. The desired time during the lunar month for arrival in lunar orbit will be determined by the time of year (Sun's location), assuming it is desired that the surface stay be during lunar daylight.

Halo Orbit—A halo orbit, with the penalty of somewhat higher mission delta V's and slow transfers, allows access to any surface site any time. Transfer times are about 8 days each way between Earth orbit and halo orbit and about 3 days each way between the halo orbit and the surface. Thus, halo orbit missions with a 14-day surface stay will be approximately 38 days in length for any site location. The halo orbit, always in view of Earth, also provides communications relay capability for continuous communication with about 90% of the lunar far side.

Direct—The direct flight mode permits free selection of lunar orbits since the lunar departure orbit is unconstrained by the lunar arrival orbit. Therefore, nominal lunar orbit wait times will be about 1 day, leading to a total 22-day mission for a 14-day surface stay and 3-day transfers.

Comparison—Figure 3.3-5 shows a comparison of the representative timelines for the modes discussed.

3.3.1.2.3 Transfer and Storage

Not applicable.

3.3.1.2.4 Orbital Assembly, Maintenance, and Modification

Not applicable.

3.3.1.3 Transportation Requirements

3.3.1.3.1 Payload Delivery Points

Payload delivery points are summarized in table 3.3-8, with vehicle utilizations for the reference transportation mode. These are generally applicable to all transportation modes.

Table 3.3-8. Independent Lunar Surface Mission Nominal Payload Delivery Points

Mission Phase \ Delivered Item	Lunar Sortie Vehicle (LSV)	Exploration and Science Equipment	Orbit Transfer Vehicle (OTV)	Crew
To Earth orbit	SS	SS	SS or HLV	SS
Earth orbit to Lunar orbit	OTV	OTV	SELF	OTV
Lunar orbit to Lunar surface	SELF	LSV	—	LSV
Lunar surface to Lunar orbit	SELF	—	—	LSV
Lunar orbit to Earth orbit	—	—	SELF	OTV
Earth landing	—	—	SS if applicable	SS
Disposal	Crash on Moon	Left on Moon	—	—

Note: SS = Space Shuttle
HLV = Heavy Lift Vehicle

3.3.1.3.2 Payload Delivery Options

Sizes (envelopes) are approximately as stated in table 3.3-9.

Table 3.3-9. Payload Size Characteristics

Item	Meters (DXL)	Feet (DXL)
CEM	4.2 x 4.7	13.8 x 15.5
EEM	4.4 x 4.3	14.5 x 14.0
Payload pallets (2 required)	4.4 x 6.0	14.5 x 20.0

A 20% mass growth allowance was applied to CEM inerts. Since the EEM was based on Apollo hardware, no growth allowance was deemed necessary. The 4 535 kg (10,000 lb) surface exploration payload was a requirement specified in the study statement of work and was assumed to include growth. Payload masses used in sizing transportation systems are presented in table 3.3-10.

Table 3.3-10. Payload Mass Characteristics

Payload Item	REO Mode		Apollo Mode		Direct Mode	
	kg	lb	kg	lb	kg	lb
Total initial payload	13 425	29,500	19 585	43,175	13 260	29,230
Landed payload	12 960	28,570	12 575	27,720	12 795	28,210
Ascent payload	7 975	17,580	7 590	16,735	7 810	17,220
Return payload	7 510	16,555	8 225	18,135	7 810	17,220

3.3.1.4 Mission/Transportation Modes and Operations

3.3.1.4.1 Transportation Options

Principal transportation candidates considered for the ILSS mission were compatible with launch by the space shuttle. There were:

- Small LO_2/LH_2 orbit transfer vehicle (OTV) staged as required with return to Earth orbit (reference mode);
- LO_2/MMH OTV, tandem staged (for equal states) for delivery to the moon and single stage for lunar descent and ascent; return to Earth orbit;
- LO_2/MMH OTV's and lander operated in the Apollo mode;
- LO_2/MMH OTV's and lander with direct lunar landing and direct Earth entry return (comparison only).

Additional candidates, not compatible with shuttle launch, were·

- Large single-stage LO_2/LH_2 OTV for Earth-moon transportation; single-stage LO_2/LH_2 lander; return to Earth orbit.
- The same modes except for use on tandem-staged LO_2/LH_2 OTV's for Earth-moon transportation.

3.3.1.4.2 Representative Transportation Mode and System

3.3.1.4.2.1 Transportation System

The reference mode vehicle assembly is depicted in Figure 3.3-6, and the mode sequence in Figure 3.3-6A. The mode as depicted results in expenditures of two drop tanks, two orbit transfer vehicles, and the lunar exploration payload. One OTV and the crew and equipment module are recovered by a shuttle flight at the end of the mission. Table 3.3-11 is a tabulation of the time, delta V and mass histories for the reference mode.

3.3.1.4.2.2 Transportation Sizing

The reference sequence was sized on a point-design basis.

3.3.1.4.2.3 Operational Factors

Operations Derived Requirements—

- Assembly of the entire mission vehicle in Earth orbit is required. Vehicle elements may be required to remain in Earth parking orbit as long as several weeks while the assembly sequence is completed. The degree of assembly is dependent on selection of Earth-to-orbit transportation. Even if a heavy-lift system is available (capable of launching the entire mission in a single launch) some orbital assembly may be required to minimize launch loads carried through mission-vehicle structure. Assembly is likely to require unmanned, but remote piloted, docking.
- Transfer of propellants from a tanker to the mission vehicle may be required, depending on the Earth-to-orbit transportation and mission transportation systems selected.
- The portion of the mission vehicle left in lunar orbit must remain serviceable unattended for the 14-day duration of the surface mission and provide a cooperative (but passive) docking target at the completion of the surface mission. Rendezvous techniques will be as used for Apollo.
- The LSV must be capable of docking with cargo modules delivered to orbit by the shuttle; it must also be capable of rotating and translating these cargo modules to a suitable location for lunar landing.
- The LSV must be able to jettison "down payload" cargo in order to accomplish a landing abort and return to lunar orbit.

Crew Involvement—The independent lunar surface mission is a manned mission; the crew is involved

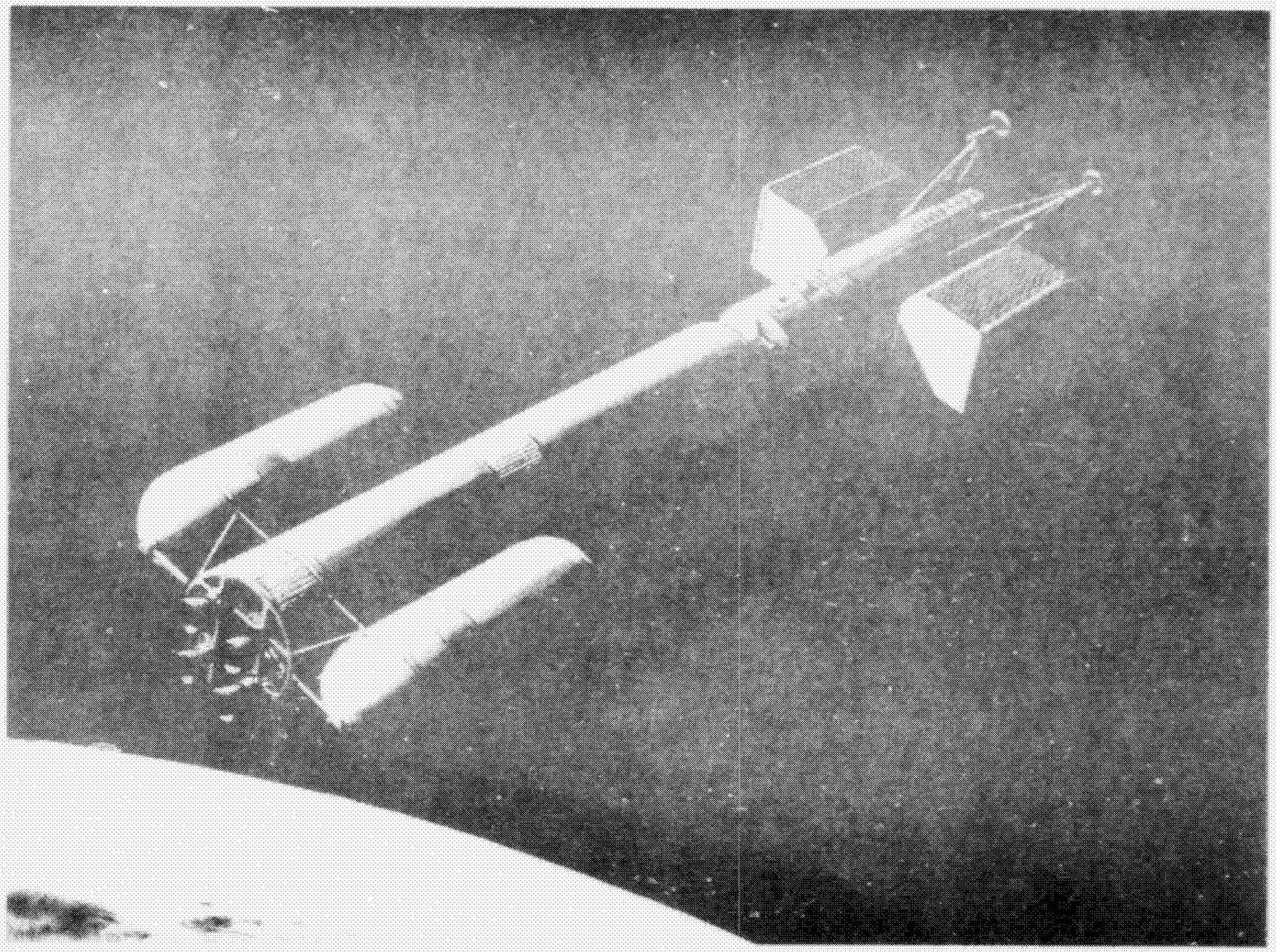
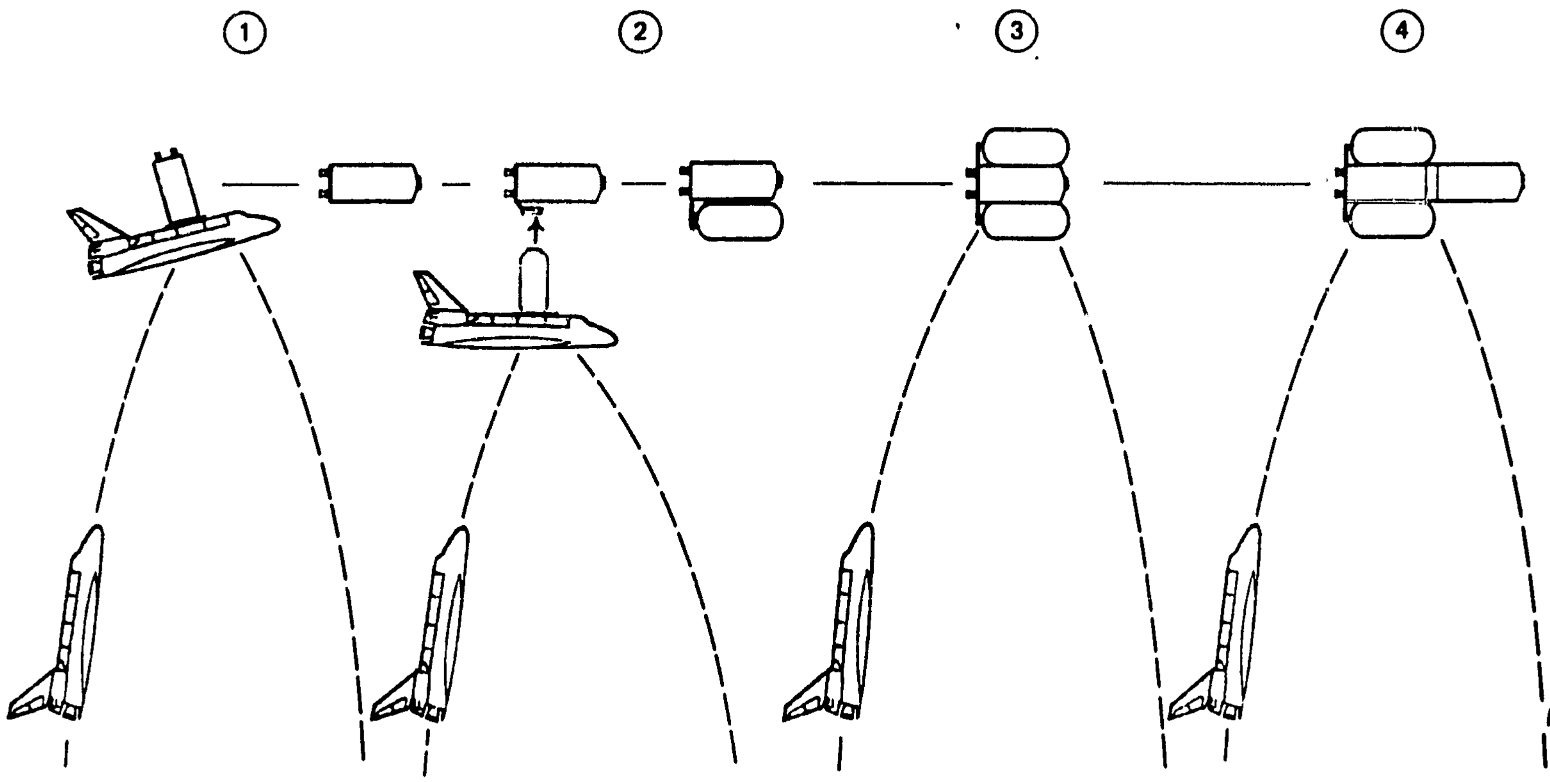


Figure 3.3-6: Clustered Small OTV Assembly for ILSS

EARTH ORBIT OPERATIONS



PHASE TIME	0	1 WEEK	1 WEEK	1 WEEK	1 WEEK
CUM TIME	0				

1	2	3	4
• DELIVER TLI STAGE WITH SS	• DELIVER DROP TANK NUMBER 1 WITH SS • ROTATE DROP TANK INTO PLACE	• DELIVER DROP TANK NUMBER 2 WITH SS • ROTATE DROP TANK INTO PLACE	• DELIVER LOI/TEI/EOI STAGE WITH SS

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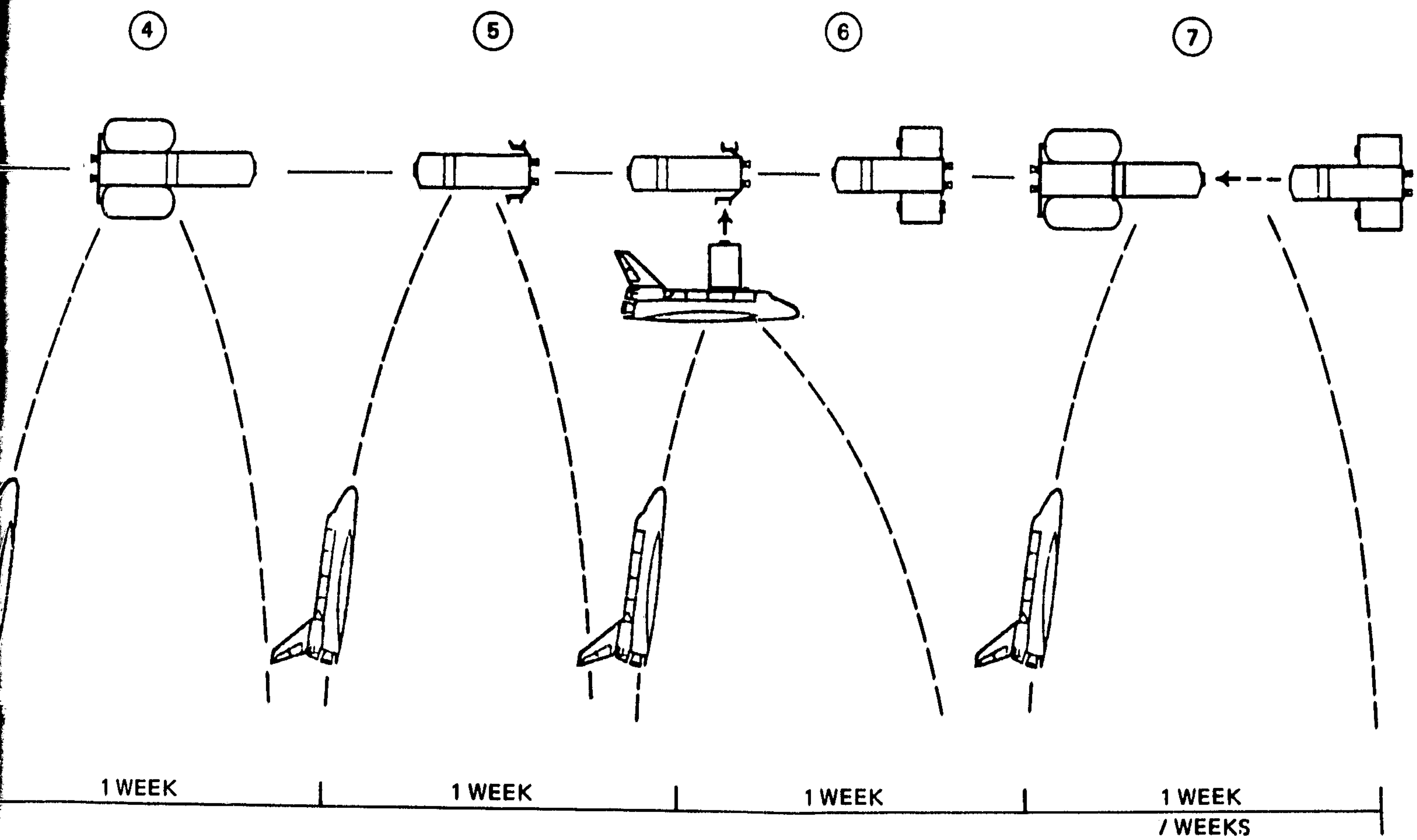
TH ORBIT OPERATIONS



4	5	6	7
• DELIVER LOI/TEI/EOI STAGE WITH SS	• DELIVER SORTIE VEHICLE WITH SS	• DELIVER SORTIE PAYLOAD MODULES WITH SS • ROTATE MODULES INTO PLACE	• DELIVER CREW AND TOP OFF PROPELLANT WITH SS • ASSEMBLE ENTIRE VEHICLE

Figure 3.3-6A ILSS Mission Transportation Sequence (Sheet 1)

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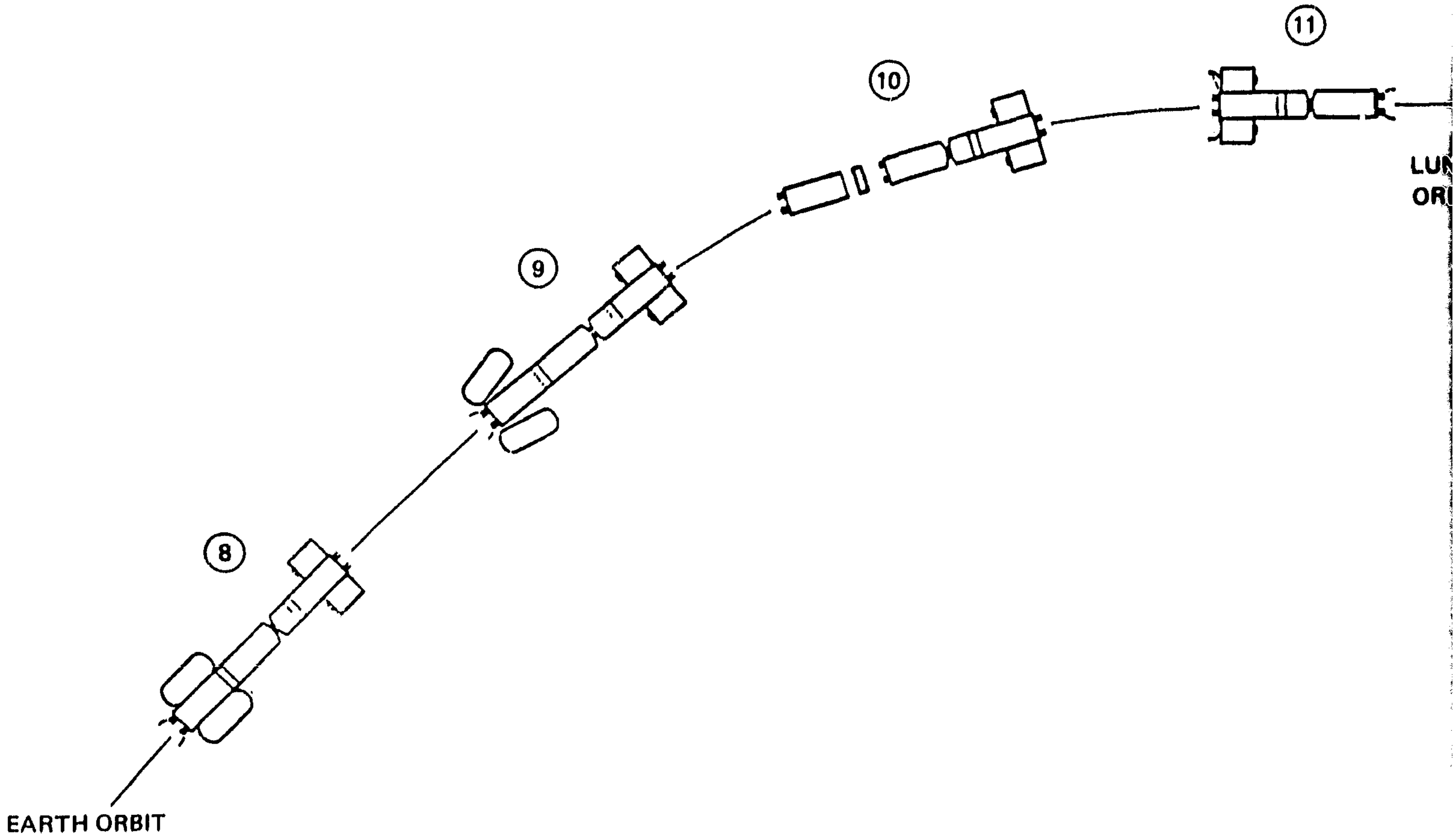
LUNAR TRANSIT AND SURFACE OPERATION



PHASE TIME 0

CUM TIME 0 .3 DAYS

8	9	10	11
• BEGIN TRANSLUNAR INJECTION	• JETTISON DROP TANKS • CONTINUE TRANSLUNAR INJECTION	• TRANSLUNAR COAST • SEPARATE TLI TUG AND DOCKING RING	• LUNAR ORBIT INSERTION

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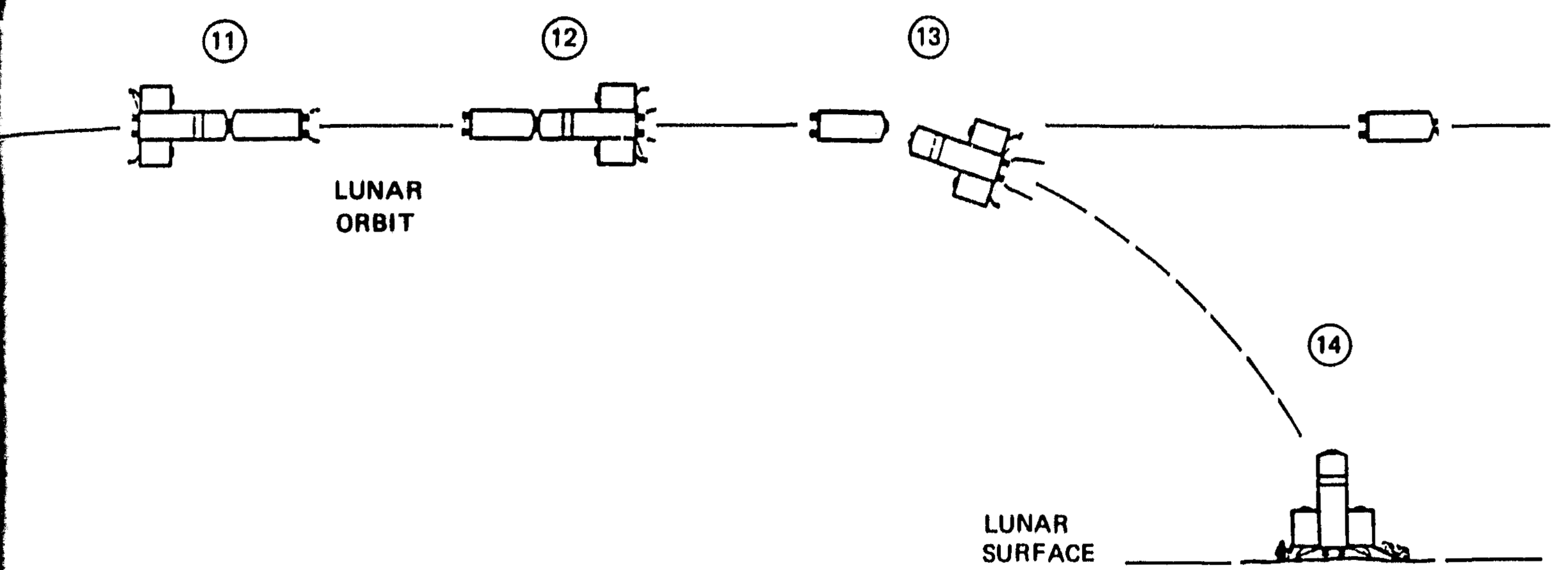
SIT AND SURFACE OPERATIONS



	11	12	13	14
Duration	3 DAYS	7 DAYS		14 DAYS; 24 DAYS
	• LUNAR ORBIT INSERTION	• LUNAR ORBIT CIRCULARIZATION • WAIT FOR LANDING SITE ALIGNMENT	• LUNAR DESCENT • TEI/EOI TUG LEFT IN LUNAR ORBIT	• LUNAR LANDING AND SURFACE MISSION

Figure 3.3-6A ILSS Mission Transportation Sequence (Sheet 2)

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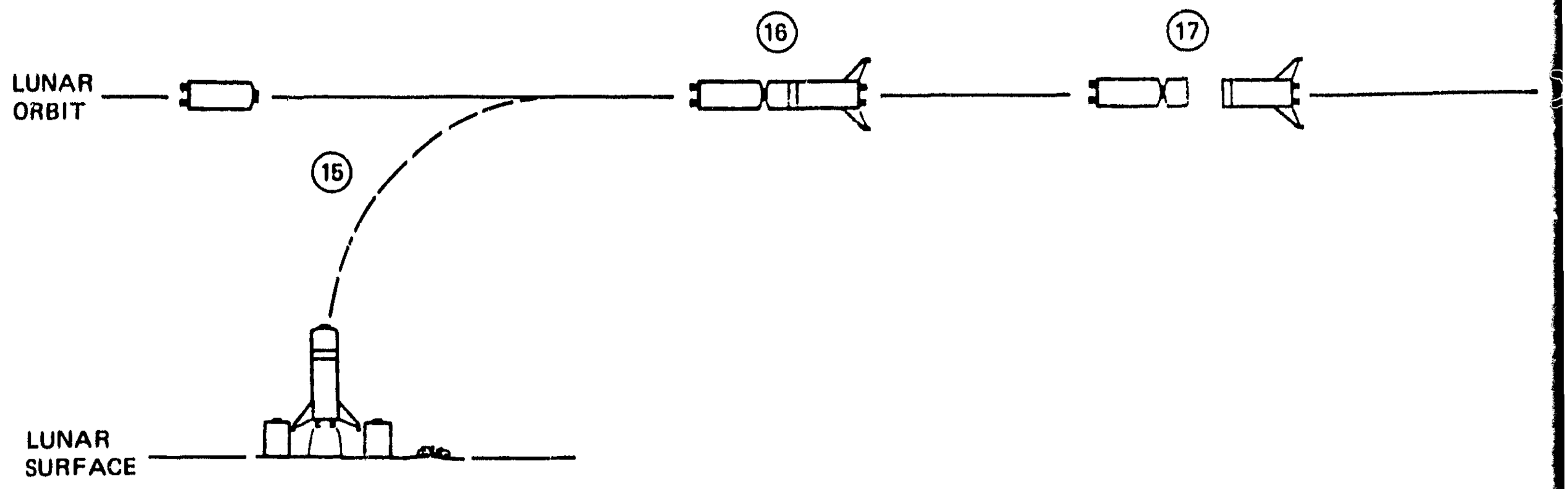
EARTH RETURN OPERATIONS



PHASE TIME	0		3 DAYS	
CUM TIME	0			
	15	16	17	
	• LUNAR LIFTOFF • CARGO MODULES LEFT ON SURFACE	• RENDEZVOUS AND DOCKING WITH TEI/EOI STAGE	• SEPARATE LANDING/ DESCENT STAGE • WAIT FOR EARTH RETURN OPPORTUNITY	• TRAN INJEC • COAS

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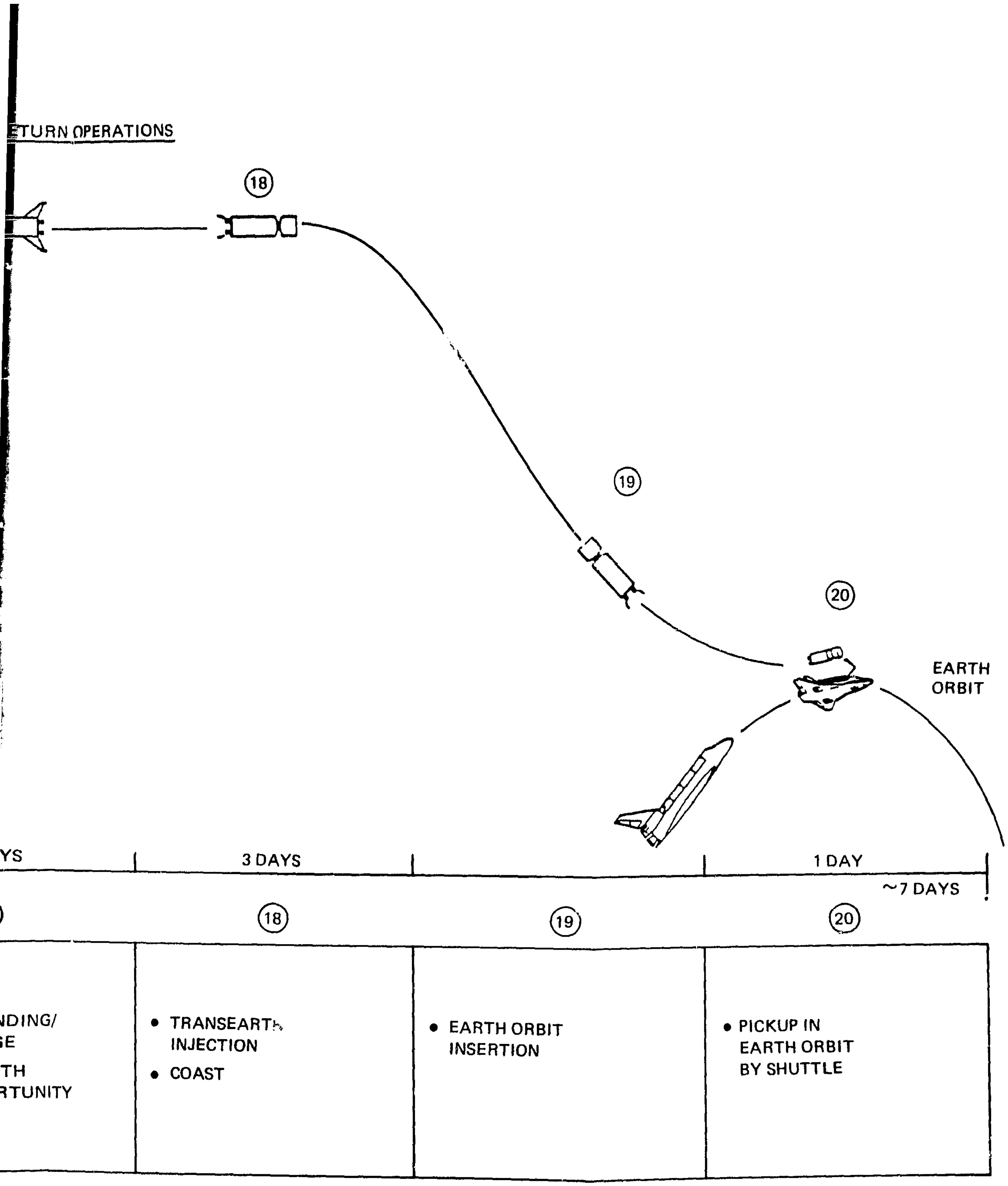

Figure 3.3-6 A ILSS Transportation Sequence (Sheet 3)

Table 3.3-11. Mission History

EVENT	CUM TIME HR	ΔV		MASS REMAINING	
		M/SEC	FT/SEC	10^3 KG	10^3 LB
INITIAL CONDITION (LEAVE LEO)	0			152.8	337.0
BOOST	0.5	2050	6725	96.5	212.7
SEPARATE DROP TANKS	0.5	—	—	92.7	204.4
INJECT (TLI)	0.6	1300	4265	69.2	152.6
DROP TLI STAGE	0.6	—	—	66.2	145.9
LOI-1 (ELLIPTIC ORBIT) (LANDING STAGE)	90.6	400	1312	59.7	131.5
LOI-2 (CIRCULARIZE) (RETURN STAGE)	94.7	525	1722	52.9	116.7
SEPARATE LANDING STAGE	95			18.3	40.4
RENDEZVOUS WITH RETURN PAYLOAD AND CREW	434.7	—	—	25.3	55.9
TEI (RETURN STAGE)	434.8	945	3100	20.3	44.8
EOI (RETURN STAGE)	545	3145	10,318	9.7	21.4
LUNAR LANDING					
INITIATE DESCENT	95			34.6	76.3
DESCENT & LANDING	97	2172	7125	21.3	97.0
READY FOR ASCENT	433	—		16.2	35.6
ASCENT	434.7	2000	6561	10.3	22.8

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in all mission phases. The lunar mission vehicle is assembled in orbit through a sequence of docking maneuvers: it is envisaged that the shuttle crew will perform the unmanned dockings flying the stages in a remote piloted mode. Two of the unmanned dockings are followed by a rotation maneuver to achieve the side-by-side configuration for the TLI stage and its drop tanks. The rotation will be assisted or performed by the crew using the shuttle-attached manipulator (SAMS).

The independent lunar surface mission is generally analogous to the Apollo lunar missions in its abort and rescue characteristics. Prior to initiation of lunar descent, either of the two stages remaining can deliver the crew module back to an Earth orbit for shuttle pickup. However, after lunar descent, the repertoire of abort and rescue modes available to the OLS and lunar surface base missions is not available. These modes could be provided by sending a group flight of two independent lunar surface mission vehicles to the moon; one vehicle would wait in lunar orbit while the other would conduct the surface mission. However, the practicality of this is doubtful. This abort/rescue situation applies to all of the candidate mission modes discussed, except that in the direct mode the backup flight—if used—would also land on the lunar surface. If a heavy-lift vehicle were available and capable of launching the entire ILS mission vehicle in a single launch, a rescue launch from Earth could be employed. Assembly of a rescue vehicle in Earth orbit is not excluded, but it would be too time consuming in most situations.

Control Functions and Requirements—All the independent lunar surface sortie missions and transportation modes involves three critical maneuvers requiring precision targeting:

- Lunar orbit insertion
- Lunar landing
- Earth return to direct entry or Earth orbit

In addition, most of the modes require lunar orbit rendezvous.

Network Support—The ILSS mission is envisaged as analogous to Apollo and would probably be flown only a few times, for example to fill in gaps in lunar science. Accordingly, it may be expected to require continuous network support as did Apollo.

3.3.1.4.2.4 Earth Launch Requirements Summary

The representative mode requires seven shuttle flights to assemble the mission vehicle in Earth orbit and one flight to retrieve the returning crew.

3.3.1.4.3 Transportation Options Comparison and Evaluation

The following variations on the reference mode were investigated:

- Recovery of the two OTV's expended in the reference mode. Referring to the sequence pictorial, after separation (step 10) the TLI stage executes a lunar free return trajectory and returns to Earth orbit. At separation it retains sufficient propellant for the Earth orbit insertion burn. In step 17, the landing stage does not separate but is returned to Earth orbit as return payload. The TEI/EOI stage is increased in size as required by the extra return payload. Three shuttle flights rather than one are required at end of mission to return the recovered hardware to Earth. This mode is about 20% heavier than the reference mode (mostly propellant) at start of mission.

- Tandem staged LO_2/MMH OTV's for Earth-moon transportation plus a single-stage LO_2/MMH lander. The lander operates the same as the reference mode; the tandem-staged OTV's operate in a manner analogous to that used for the geosynchronous satellite maintenance sortie. The tandem stages return to Earth orbit for reuse. The lander may be reused or expended at the moon after ascent. The mass penalty, in terms of initial mass in Earth orbit, for lander reuse is roughly 40 000 kg (88,000 lb).

- LO_2/MMH stages in an Apollo-LOR mode (single-stage lander). The mode is the same as that used for Apollo except that the crew resides in the crew and equipment module (used for the lunar surface sortie) during the entire mission except Earth return.

- LO_2/MMH stages in a direct landing/direct return mode. After arrival in lunar orbit, the entire remaining vehicle (lander, ascent/TEI stage, and Earth entry module) lands on the moon. After surface mission completion, the ascent/TEI stage lifts off and delivers the EEM to lunar orbit and then to an Earth return (direct entry) trajectory. As noted earlier, this mode does not provide an adequate crew habitable volume as defined. It is not recommended for further consideration.

- LO_2/LH_2 single-stage OTV with a LO_2/LH_2 single-stage lander, all stages reused. The single-stage OTV waits in lunar orbit for the surface sortie. After the surface mission the OTV returns the lander crew and equipment module, and crew to Earth. The single-stage OTV is not compatible with shuttle launch.

- LO_2/LH_2 tandem staged (two equal stages) OTV with LO_2/LH_2 single-stage lander. The mission is the same as above except that the tandem stage boost element is separately recovered.

3.3.1.4.3.1 Size and Performance Comparison

Those transportation modes that appeared primarily suited only to this mission were point-matched. Those applicable to other lunar missions (orbiting lunar station and lunar surface base) were parametrically matched. Figure 3.3-7 compares masses of the alternatives modes and figure 3.3-8 compares sizes. Figures 3.3-9 through 3.3-12 present vehicle and stage matching/sizing data for most of these modes.

3.3.1.4.3.2 Earth Launch Transportation Comparison

The number of shuttle and/or heavy-lift flights required to accomplish the ILSS options are summarized in figure 3.3-13.

3.3.1.4.3.3 Operational Comparison

Operational factors are summarized in table 3.3-12. The common stage modes require tracking and control of two independent vehicles. For the clustered small OTV mode, if the TLI stage is to be recovered, it must be tracked and controlled on a lunar free return while the lunar mission is in progress.

3.3.1.4.3.4 Practicality Assessment

All of the modes analyzed appear to be practical. Maximizing recoverability of the representative mode does not seem worthwhile--the additional shuttle flights required will largely negate recovery savings and mission operations are considerably more complex. Table 3.3-13 presents pro's and con's for the alternatives.

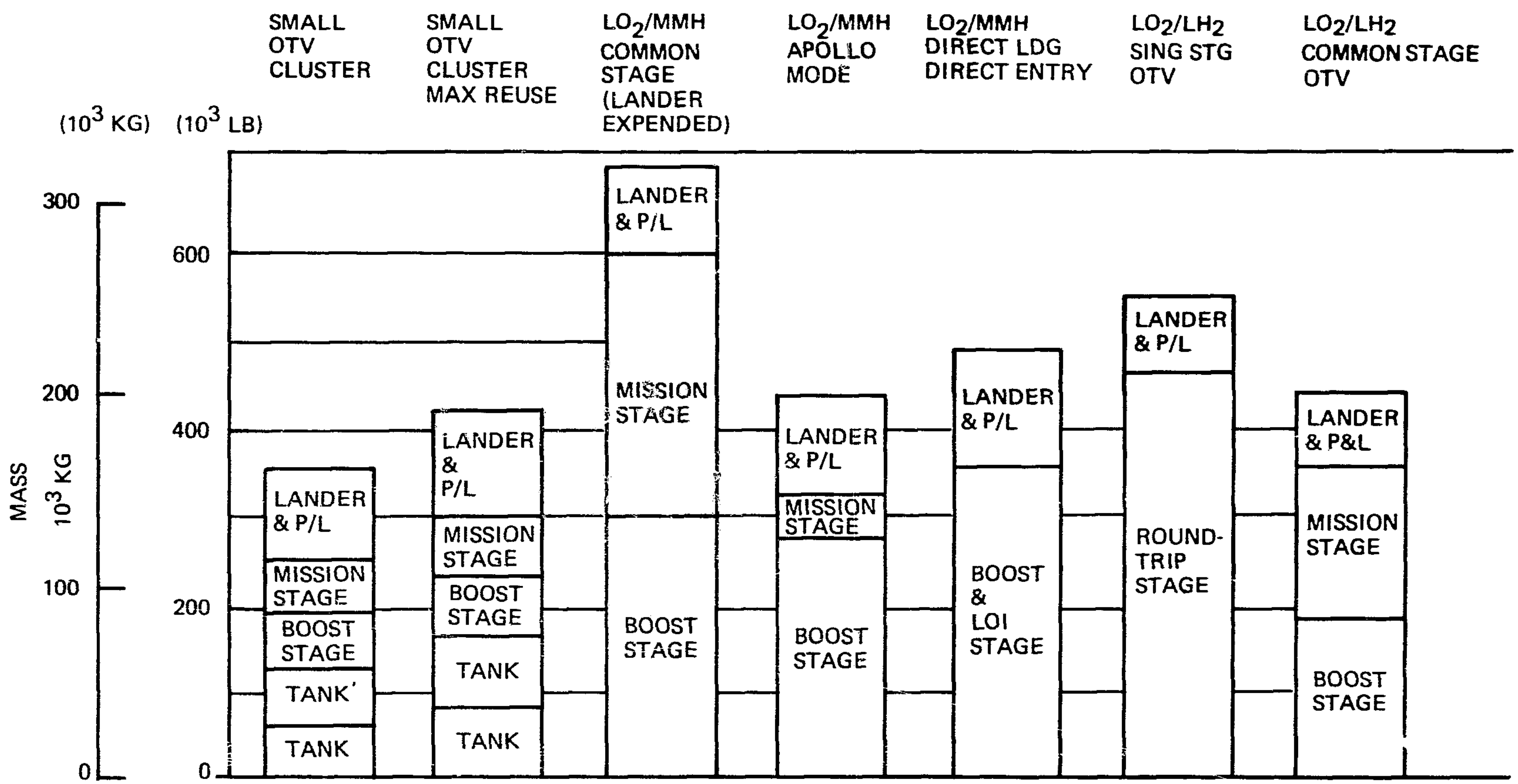

Figure 3.3-7 Mass Comparison of ILSS Options

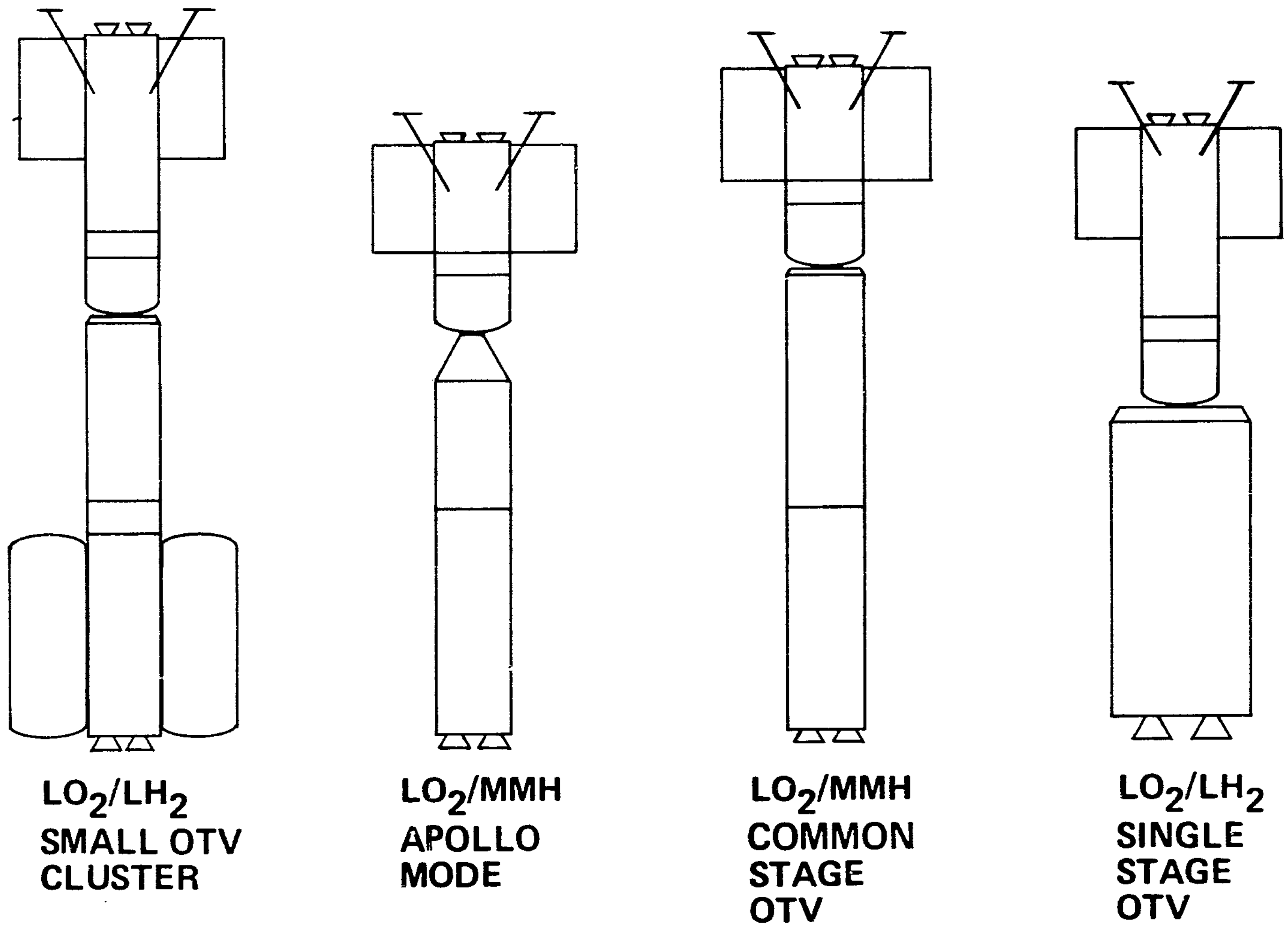

Figure 3.3-8. Size Comparison of Representative ILSS Transportation Modes

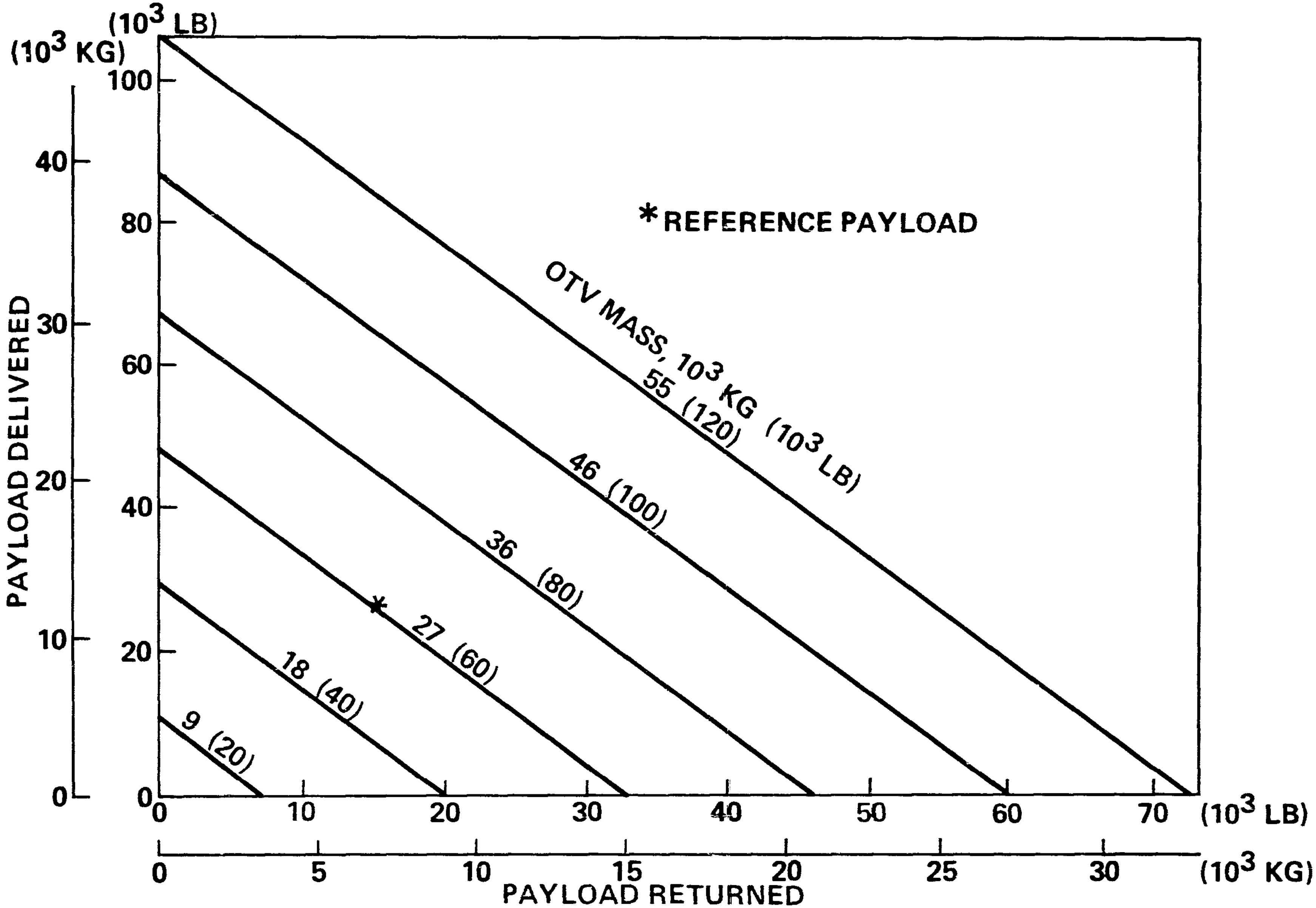

Figure 3.3-9. Single Stage LO_2/LH_2 LSV Capability for ILSS

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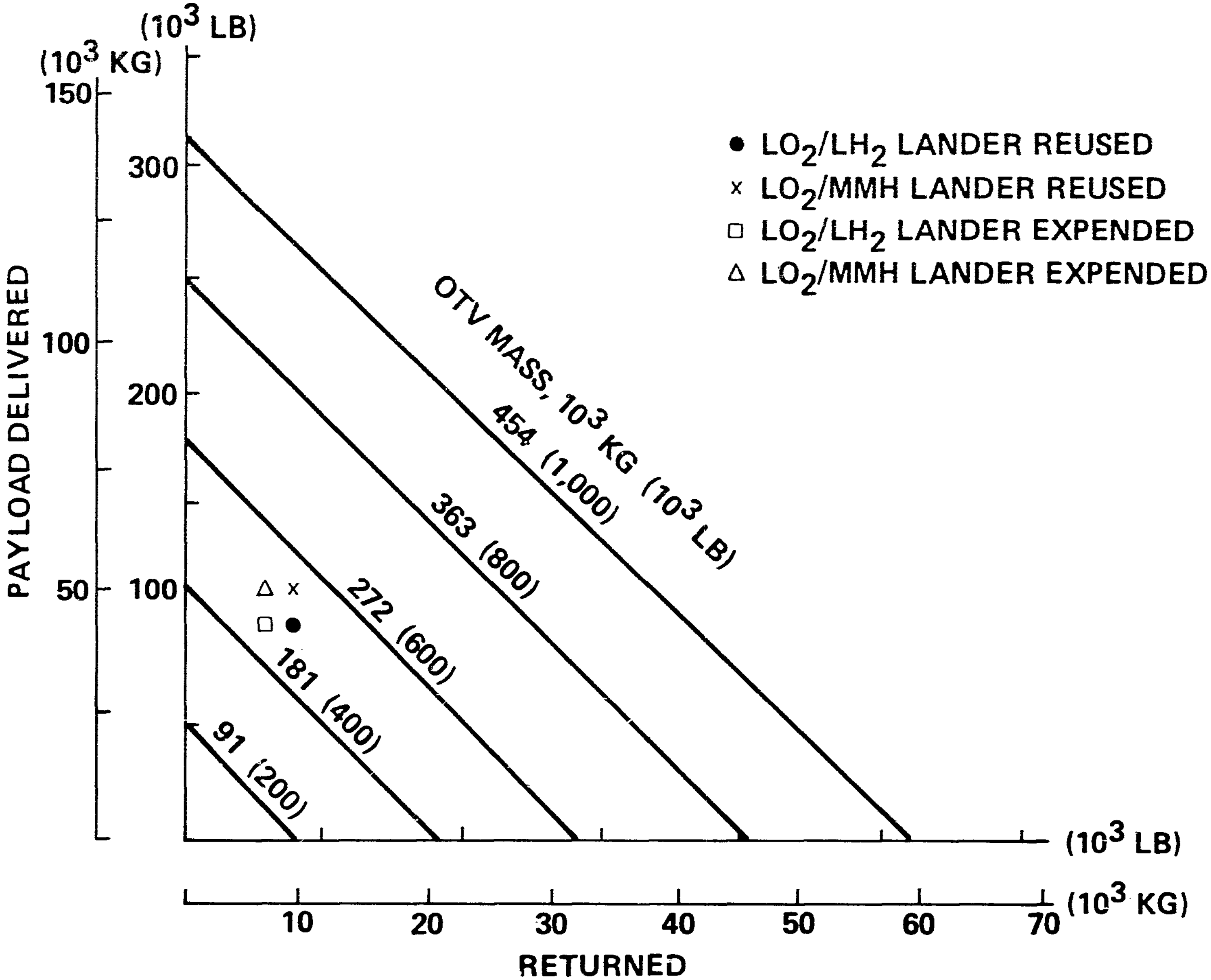


Figure 3.3-10. Single Stage LO_2/LH_2 OTV Capability for ILSS

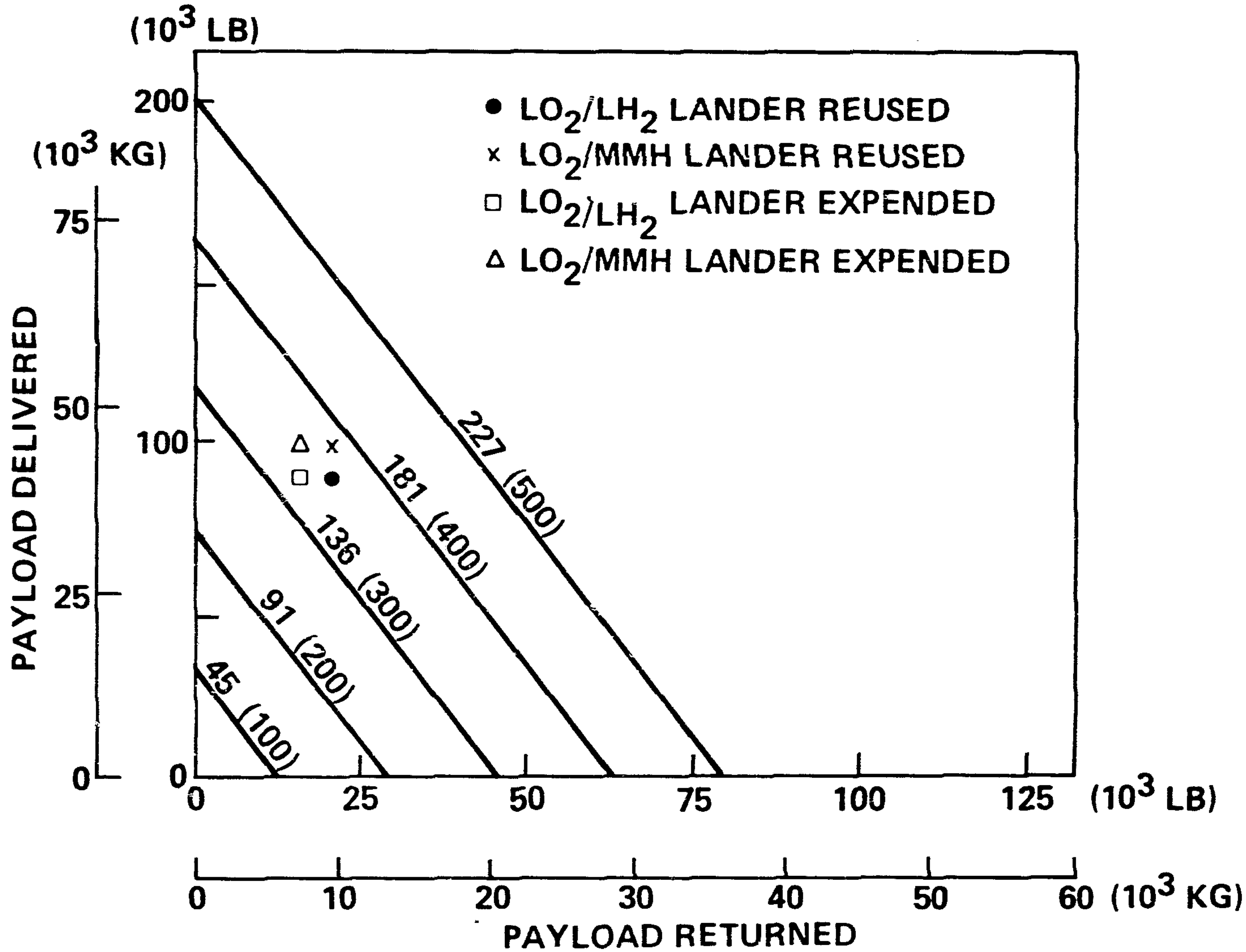


Figure 3.3-11: Common Stage LO_2/LH_2 OTV Capability for ILSS (Return to Earth Orbit)

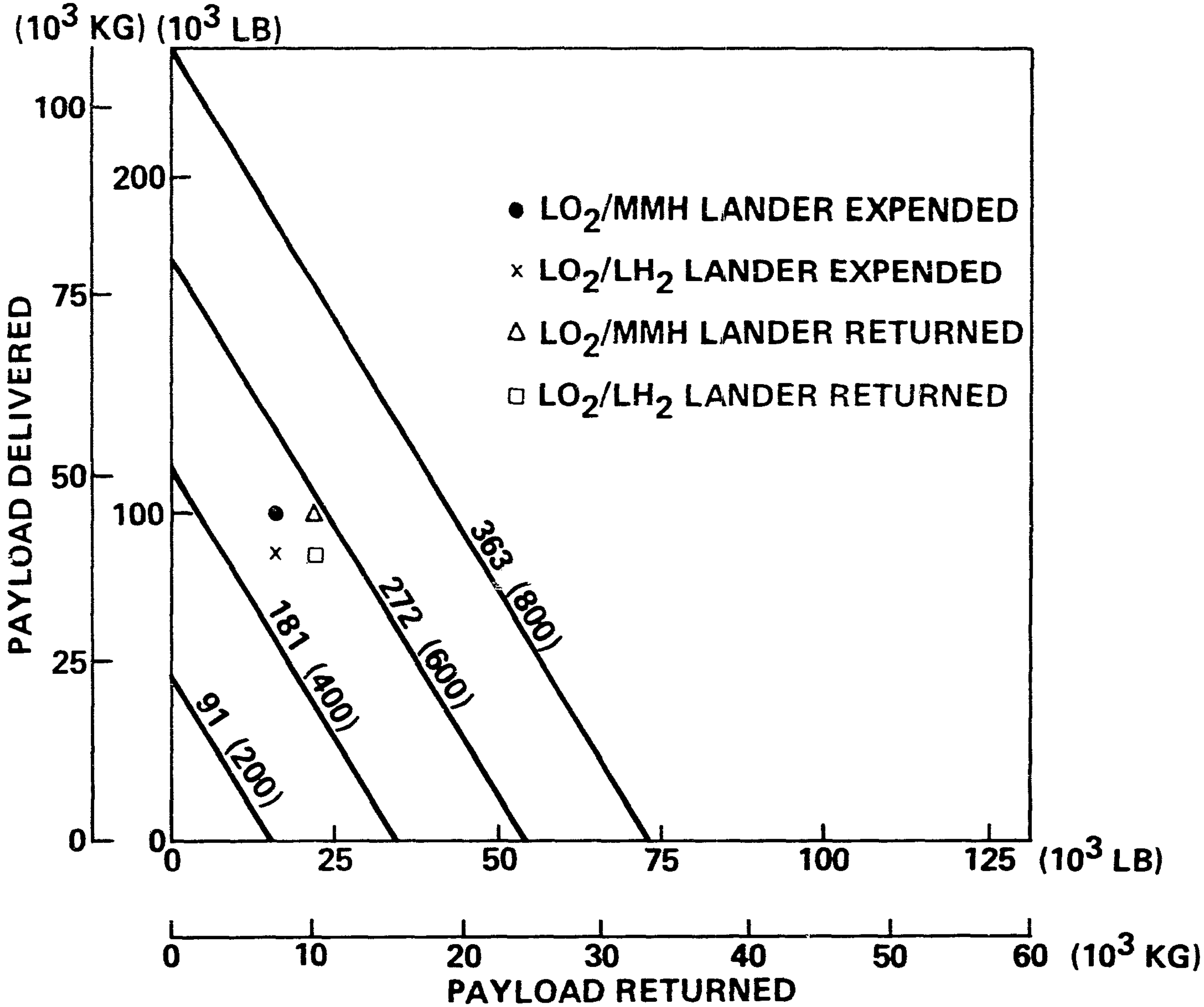


Figure 3.3-12: Common Stage LO_2/MMH OTV Capability for ILSS (Return to Earth Orbit)

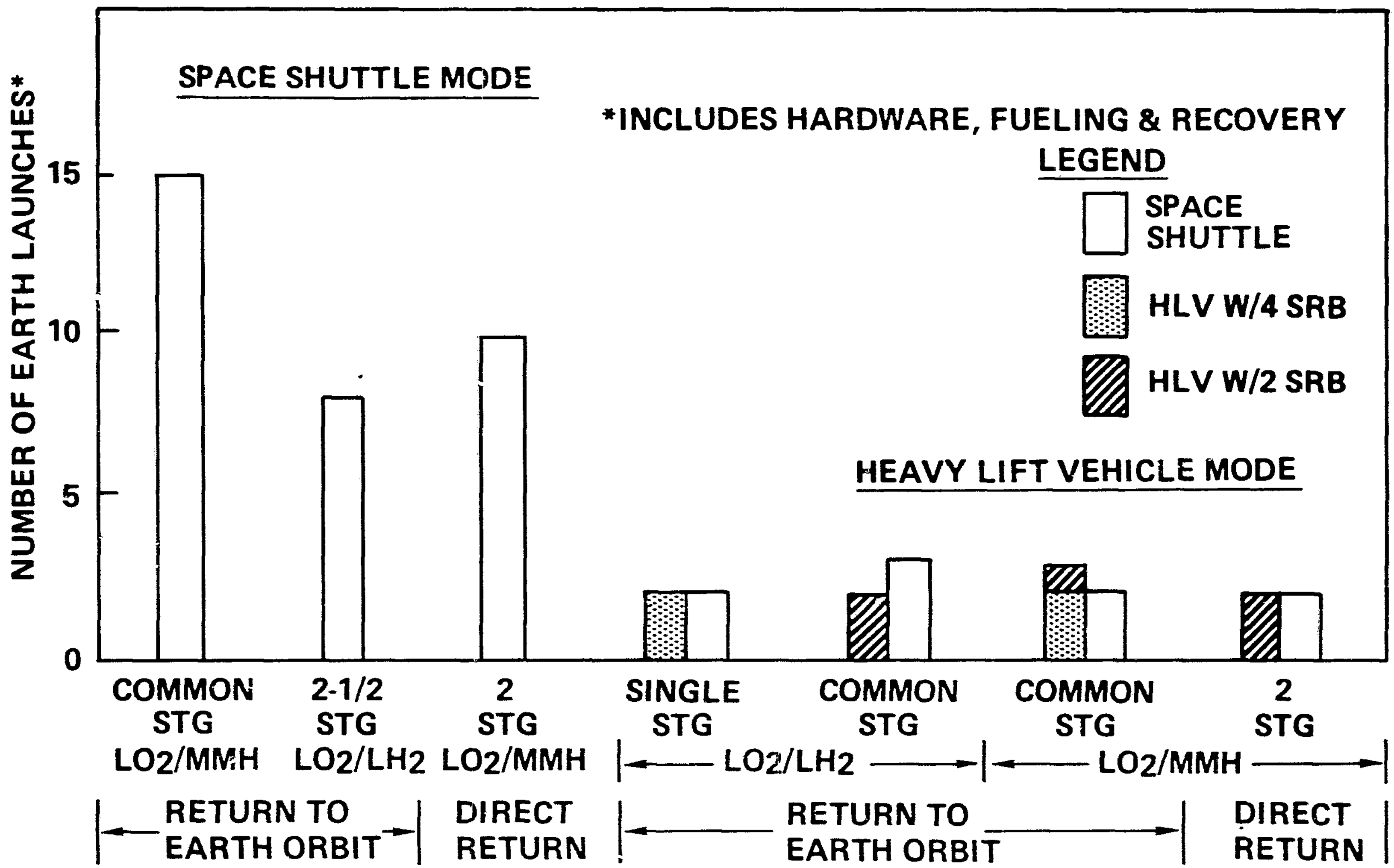


Figure 3.3-13. Earth Launches Required for One ILSS Mission

Table 3.3-12. Operational Factors Summary

	Mode \ Function	Navigation/targeting: Lunar orbit insertion	Navigation/targeting: Lunar landing	Navigation/targeting: Lunar orbit rendezvous	Navigation/targeting: Earth return	Docking: Head-to-head Apollo /Soyuz type	Docking: Side by side (drop tanks)	Docking: Large diameter interstages	Fueling/ refueling	Shuttle retrieval recovery	Sea recovery
Potentially compatible with shuttle	Clustered small OTV (reference)	●	●	●	Orbit	●	●	●	●	●	
	LO_2/MMH common stage	●	●	●	Orbit	●		●	* 136,000 kg (300,000 lb)	●	
	LO_2/MMH Apollo-mode	●	●	●	Direct	●		●	* 122,500 kg (270,000 lb)		●
	LO_2/MMH direct landing & direct return	●	●		Direct	●		●	* 160,000 kg (352,000 lb)		●
Requires heavy lift	LO_2/LH_2 large OTV (single stage)	●	●	●	Orbit	●			●	●	
	LO_2/LH_2 large OTV common stage	●	●	●	Orbit	●		●	**	●	

* Fueling can be avoided if a heavy lift vehicle of sufficient capability is avoided. Capabilities required are indicated

** Can be launched fueled by heavy lift vehicle. OTV reuse requires refueling

Table 3.3-13. ILSS Mode Assessment

	ADVANTAGES						DISADVANTAGES			
	COMPATIBLE WITH SHUTTLE LAUNCH AND RECOVERY	FULLY REUSABLE	ONLY ONE STAGE TO DEVELOP	ONLY ONE MISSION SPACECRAFT TO DEVELOP	NO LH_2	NO LUNAR ORBIT RENDEZVOUS	GREAT MASS REQUIRED	MULTIPLE STAGE-TO-STAGE DOCKING	TWO ACTIVE VEHICLES TO TRACK	INADEQUATE CREW VOLUME
SMALL OTV MULTISTAGED	x		x	x				x		
SMALL OTV MULTISTAGED & MAX REUSE	x	EXCEPT DROP TANKS	x	x				x	x	
LO_2/MMH COMMON STAGE	x	x		x	x		x	x	x	
LO_2/MMH APOLLO MODE	x				x			x		
LO_2/MMH DIRECT LDG DIRECT RETURN	x			x	x	x		x		x
LO_2/LH_2 SINGLE STG		x		x						
LO_2/LH_2 COMMON STAGE		x		x				x	x	

3.4 ORBITING LUNAR STATION PROGRAM

A more extensive investigation of moon than that possible through the independent lunar surface sortie program could employ an orbiting lunar station as shown in figure 3.4-1. The objectives of this program would be as follows:

- Perform a broad spectrum observation of the lunar surface.
- Conduct surface sorties.
- Support and control unmanned orbital and surface operations

The orbiting lunar station program includes an orbiting lunar station (OLS) and vehicles to perform surface sorties. The principal reference on this mission is the Rockwell OLS study conducted in 1970-71 (contract NAS9-10924). The requirements determined for this mission are based primarily on this reference.

3.4.1 ORBITING LUNAR STATION MISSION

3.4.1.1 Mission Summary

3.4.1.1.1 General Description

The OLS mission employs an 8-man station as illustrated in figure 3.4-1 and operates in a lunar polar orbit at 111 km (60 n mi) altitude. A lunar halo orbit is a potential alternate location. Docked to the station are two lunar sortie vehicles each capable of a round trip to the lunar surface with a crew of four and their exploration and support equipment for a 28-day nominal surface stay. Two to four surface sorties per year are conducted. For this study, the OLS is considered to be composed of station modules derived from the modular Earth orbit space station, with the optional alternative of a unitary (single module) station derived from a similar unitary Earth orbit space station element.

The OLS and its surface sortie missions are supported by logistics flights, nominally at 109-day intervals and originating from Earth.

3.4.1.1.2 Mission Assumptions and Constraints

Nominal mission assumptions and constraints are summarized in table 3.4-1. The lunar polar orbit was selected by the Rockwell study after considerable investigation of alternatives and is adopted here. The "halo orbit" as a location for an orbiting lunar station was described by Farquhar and was briefly analyzed in this study. Depending on selection of the transportation system, the halo orbit may present advantages.

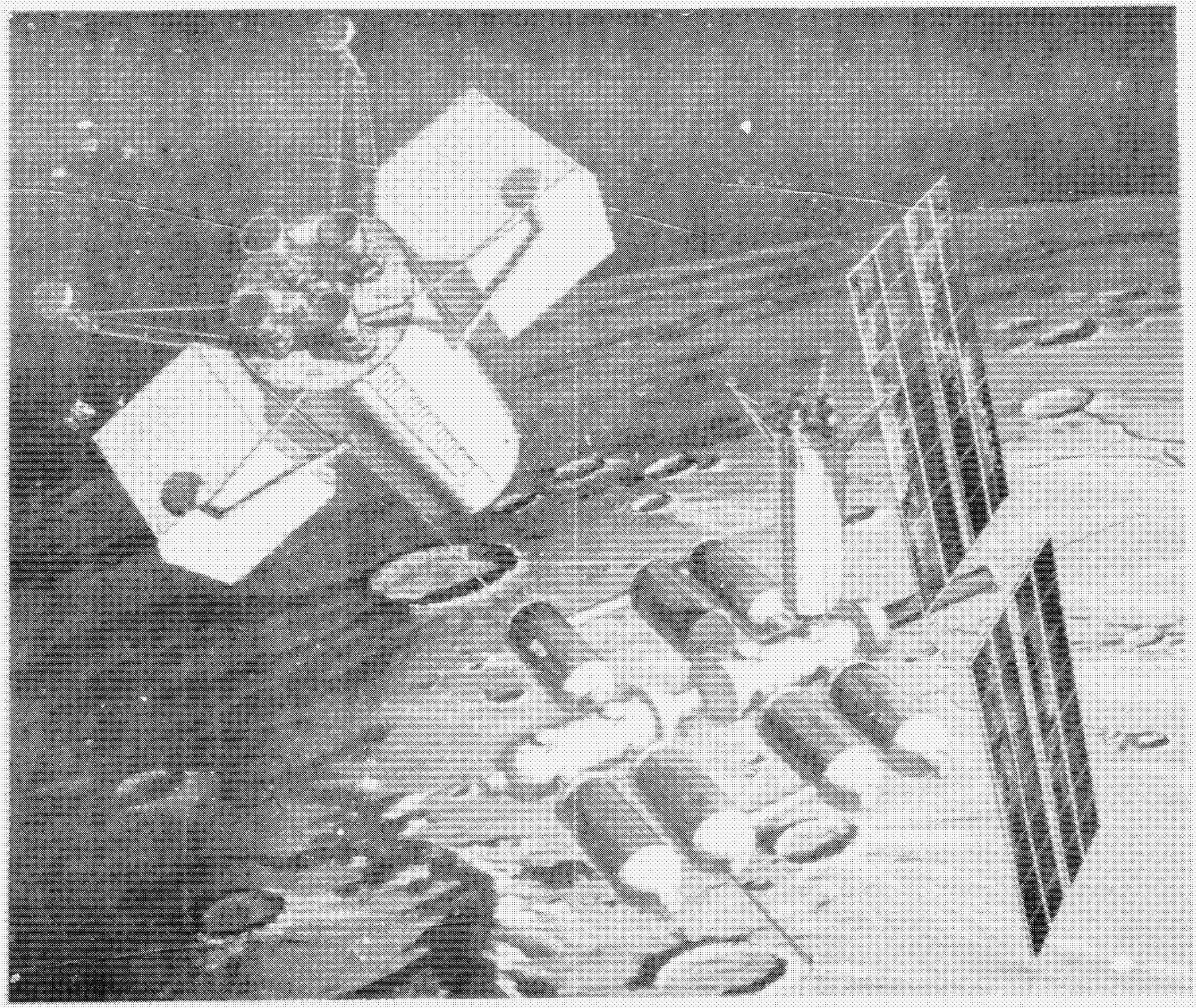


Figure 3.4-1 Orbiting Lunar Station (8 Man)

Table 3.4-1. Orbiting Lunar Station Mission Assumptions and Constraints

OBJECTIVES	ASSUMPTIONS AND CONSTRAINTS
• BROAD SPECTRUM OBSERVATION OF THE LUNAR SURFACE	• 111 km LUNAR POLAR ORBIT
• CONDUCT SURFACE SORTIES	• UP TO 8-MAN STATION CREW
• SUPPORT AND CONTROL OF UNMANNED SURFACE OPERATIONS	• ONE-MONTH DURATION 4-MAN SURFACE SORTIES UP TO FOUR TIMES A YEAR
	• DOCKING FACILITIES TO ACCOMMODATE AD HOC MODULES FOR MISSION ADAPTABILITY, LOGISTICS (RESUPPLY) MODULES, LUNAR SURFACE SORTIE VEHICLES
	• CREW EVA CAPABILITY

3.4.1.2 Mission System Description

3.4.1.2.1 Mission Options

3.4.1.2.1.1 Modular Orbiting Lunar Station (OLS)

The flight configuration for the modular OLS is shown in figure 3.4-2. Eleven modules form the basic station for a crew of eight, subsystems, experiment equipment, and consumables. Two lunar sortie vehicles (LSV's) provide capability to conduct 28-day surface explorations. Shown attached to one LSV are two modules that house mobility vehicles used on a lunar sortie. One of the LSV's is used to transfer logistics modules from the logistics space vehicle to the OLS.

Other modules such as fluid module (FM), resupply module (RM), and crew transfer vehicle (CTV) are attached only during resupply and crew rotation.

The weight of the basic station is 102 000 kg (225,000 lb) with the total OLS weight including LSV's being approximately 185 000 kg (400,000 lb).

3.4.1.2.1.2 Unitary Orbiting Lunar Station

The flight configuration for the unitary OLS is shown in figure 3.4-3. The major difference between this concept and the modular is that essentially a single module provides the required volume for the crew of eight, subsystems, experiment equipment, and consumables. All other features (including lunar sortie vehicles, operations, crew rotation, and resupply provisions) are the same.

The total orbital mass of the unitary OLS with 2 LSV's and experiment module is approximately 144,000 kg (317,000 lb) as identified in the reference.

3.4.1.2.2 Payload Descriptions

3.4.1.2.2.1 Modular OLS

A brief description of the size and weights of the 11 modules used to form the basic station is presented in table 3.4-2. Two modules (CM 1 and 2) serve as the keel to which all other modules are attached; these modules also house basic OLS subsystems. An electric power module (EPM) supports the solar array and houses cryogenics. Cryogenics are also stored in two other modules—CCM 1 and 2. These modules also serve as storage areas for bulk cargo. Two control center modules (CCM 1 and 2) are provided; one also contains the radiation shelter. Staterooms for the crew are provided in two modules (CQM 1 and 2). One module (GM) provides galley, dining, and recreation facilities. The SM contains onboard sensors and subsatellites. All modules have a diameter of 4.3 meters (14 ft) and vary in length from 8.0 to 12.8 meters (26 ft to 42 ft); these

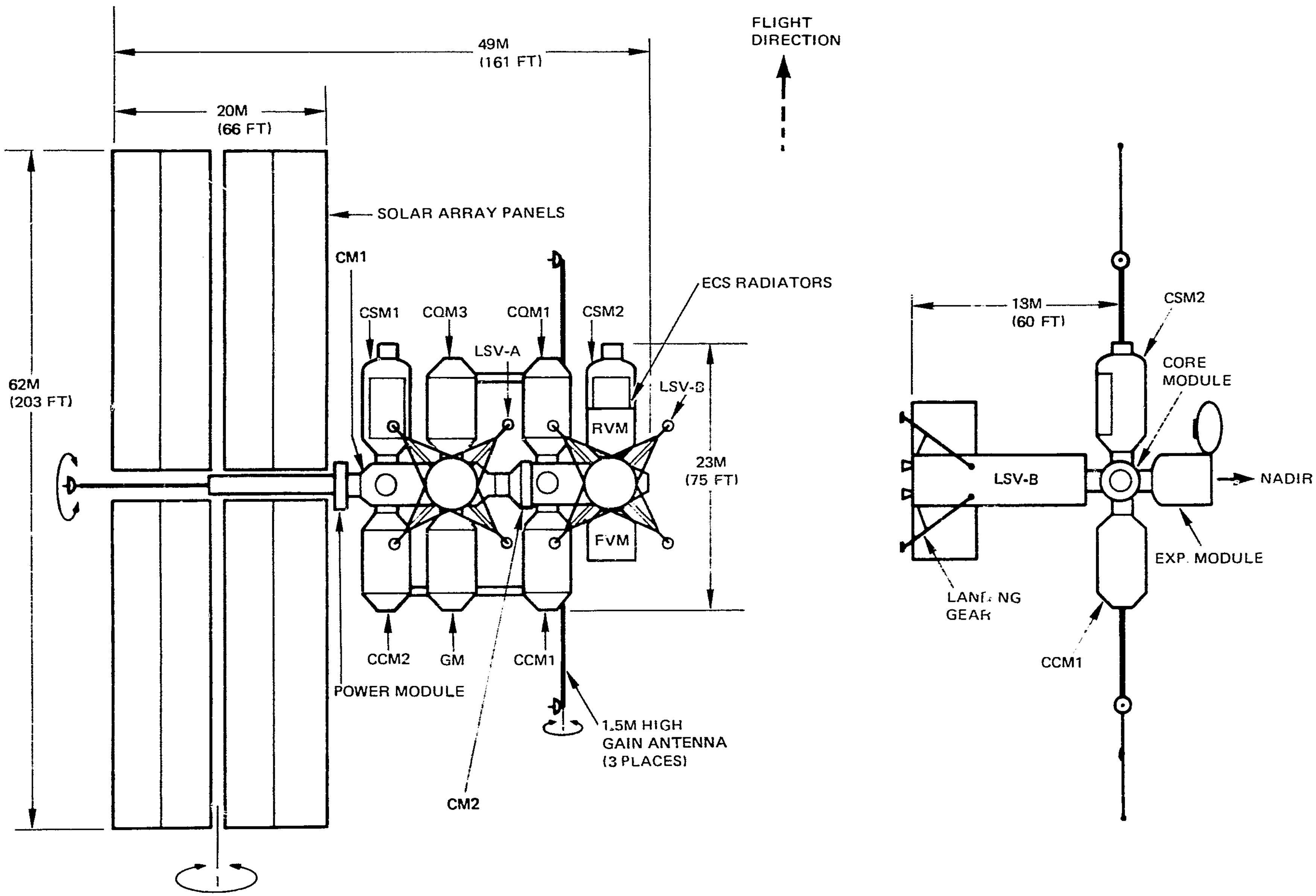


Figure 3.4-2. OLS Flight Configuration

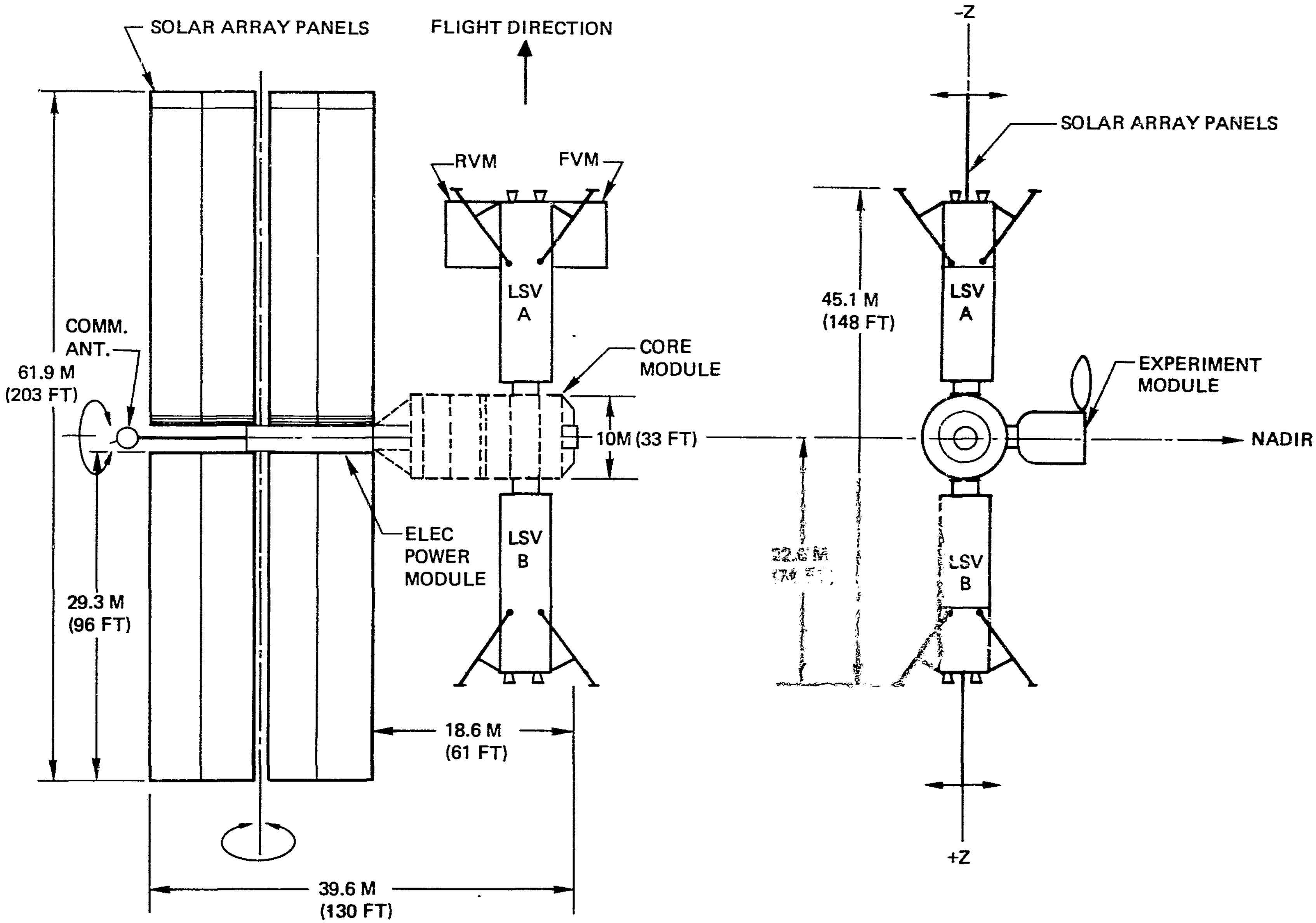


Figure 3.4-3. Unitary Orbiting Lunar Station

Table 3.4-2. Orbiting Lunar Station Modules

MODULES	DESCRIPTION	MASS		SIZE (DIAMETER x LENGTH)	
		10^3 KG	10^3 LB	METERS	FEET
1. CORE MODULE-1 (CM-1)	CENTRAL KEEL BASIC STATION SUBSYSTEMS IVA/EVA AIRLOCK	14.5	32	4.3 x 12.8	14 x 42
2. CORE MODULE-2 (CM-2)	GENERALLY SAME AS CM1	11.3	25	4.3 x 12.8	14 x 42
3. ELECTRICAL POWER MODULE (EPM)	SOLAR ARRAY CRYO STORAGE	10.9	24	4.3 x 12.8	14 x 42
4. CRYO STORAGE MODULE-1 (CSM-1)	CRYO STORAGE CARGO STORAGE	4.5	10	4.3 x 9.8	14 x 32
5. CRYO STORAGE MODULE-2 (CSM-2)	CRYO STORAGE HYRAZINE STORAGE CARGO STORAGE	12.2	27	4.3 x 9.8	14 x 32
6. CONTROL CENTER MODULE-1 (CCM-1)	BACKUP CONTROL CENTER RADIATION SHELTER LABS	16.8	37	4.3 x 9.8	14 x 32
7. CONTROL CENTER MODULE-2 (CCM-2)	PRIMARY CONTROL CENTER EXERCISE/MEDICAL	5.4	12	4.3 x 9.8	14 x 32
8. CREW QUARTERS MODULE-1 (CQM-1)	4 STATE ROOMS ECLSS FOR 8 MEN BACKUP GALLEY	6.8	15	4.3 x 9.8	14 x 32
9. CREW QUARTERS MODULE-3 (CQM-3)	GENERALLY SAME AS CQM-1 INCLUDES COMMANDERS STATEROOM	5.9	13	4.3 x 9.8	14 x 32
10. GALLEY MODULE (GM)	GALLEY AND DINING RECREATION LAUNDRY	7.3	16	4.3 x 9.8	14 x 32
11. EXPERIMENT MODULE	ONBOARD EXPER. SENSORS SUBSATELLITE M&R	5	11	4.3 x 8.0	14 x 26
25 000 KG (55,000 LB)	TOTALS	100.6	222		

*INCLUDES CONSUMABLES

dimensions are compatible with the space shuttle's cargo bay. Masses vary from 500 kg (1,100 lb) to 16 800 kg (37,000 lb). Total mass of the 11 modules as delivered includes a total of 25 000 kg (55,000 lb) of consumables.

3.4.1.2.2.2 Unitary OLS

The basic unitary station for the lunar orbit mission consists of a core module, power module, and experiment module.

The core module provides quarters for the eight-man crew and houses the majority of the subsystems. The module is divided into two separate pressure compartments for safety reasons. Each compartment consists of two transverse decks. One of the pressure compartments includes a deck for experiments and another serves as a combination crew quarters and station control deck. The second pressure compartment also includes a crew/control deck and a deck to provide galley, dining, recreation, and medical facilities. The power module supports the $930m^2$ ($10,000\text{-}ft^2$) array and houses the secondary and emergency power systems. Also supported from the module is the Earth communications antenna. The experiment module (XM) contains onboard sensors and subsatellites.

Masses and sizes of the modules are summarized in table 3.4-3.

3.4.1.2.2.3 Surface Exploration (Sortie) Payloads

The payloads for the lunar sortie vehicle (LSV) are summarized in table 3.4-4. The crew and equipment module is as defined for the ILSS mission (para. 3.3.1.2.2.1).

Surface exploration will include use of both a two-man rover and a two-man flying vehicle. Each type of mobility unit is provided with a separate storage and transport module and both will be returned to the OLS.

3.4.1.2.2.4 Crew Rotation and Resupply Payloads

Included are a crew transport vehicle (CTV), resupply module (RM), and fluids module (FM).

Crew Transport Vehicle (CTV)—The primary function of the CTV is to provide quarters for the crew during transits between Earth and lunar orbit. The CTV consists of a module providing shirt-sleeve environment for the crew and a portion of the required supporting equipment. An unpressurized equipment module is included for the remainder of the support equipment.

Table 3.4-3. Unitary OLS Mass Summary

MODULE	DESCRIPTION	MASS		SIZE (DIAMETER x LENGTH)	
		10^3 KG	10^3 LB	METERS	FEET
CORE	CREW, SUBSYSTEMS AND LABS	60.7*	133.8*	8.23 x 18.6	27 x 61
POWER	SOLAR ARRAY & POWER EQUIPMENT	11.0**	24.4**	2.28 x 12.2	7.5 x 40
EXPERIMENT	ONBOARD SENSORS & SUBSATELLITES	5	11	4.3 x 8	14 x 26
	TOTALS	76.7	169.2		

*INCLUDES 22 100 KG (48,500 LB) CONSUMABLES

Table 3.4-4. Lunar Sortie Vehicle Payloads

PAYLOADS	DESCRIPTION	MASS*		SIZE (DIAMETER x LENGTH)	
		10^3 KG	10^3 LB	METERS	FEET
1. ROVER VEHICLE MODULE (RVM)	ROVER + CONTAINER	3.2	7	4.3 x 5.2	14 x 17
2. FLYING VEHICLE MODULE (FVM)	FLYING VEHICLES (2), EXPERIMENTS, CONTAINER	3.6	8	4.3 x 5.2	14 x 17
3. CREW & EQUIP MODULE	CREW, QUARTERS & SUPPORT S/S	7.1	15.7	4.3 x 4.6	14 x 15
TOTAL LANDED		13.9	31.7		
1. RVM	EMPTY RM	2.4	5.3	4.3 x 5.2	14 x 17
2. FVM	EMPTY FV EXP'T + SURFACE SAMPLE + CONTAINER	2.3	5.1	4.3 x 5.2	14 x 17
3. CREW & EQUIPMENT MODULE	SAME AS 3 ABOVE LESS CONSUMABLES	5.8	12.8	4.3 x 4.6	14 x 15
	TOTAL RETURNED	10.5	23.2		

*EXCLUDING GROWTH

The entire crew (8 men) is normally exchanged on each crew rotation/resupply flight at 109-day intervals. The interval is dictated by lunar flight mechanics constraints.

The equipment module houses electrical power sources, cryogenic tankage, and a small emergency propulsion system capable of a 400 m/sec (1,300 ft/sec) delta V. The purpose of the propulsion system is to provide the capability for the CTV to achieve a lunar orbit or 1-day Earth parking orbit in the event the OTV fails to operate.

Fluid Module (FM)—The FM provides fluids to completely replenish one LSV, all lunar mobility vehicles, and cryogenics for the OLS atmosphere. This module is disposed of following resupply by attaching it to the OTV and jettisoning it during the return to Earth. Although the size of the FM is compatible with the space shuttle, it must be launched offloaded so that the shuttle's payload capability is not exceeded. Sizing of the fluids module depends on propellant mass required for the LSV.

Resupply Module (RM)—The RM is a pressurized container that includes bulk cargo (e.g., food, clothes, etc.) for both OLS and LSV sorties. This module is disposed of in the same manner as the FM.

Table 3.4-5 summarizes crew rotation and resupply payloads.

3.4.1.2.2.5 Consumables

Consumables are summarized in table 3.4-6. Consumables for the OLS itself are detailed in table 3.4-7.

3.4.1.2.2.6 Mass Summary

Table 3.4-8 presents a mass summary for the OLS mission options. Values do not include growth allowances.

3.4.1.2.2.7 Pickup Points and Transportation Constraints

Modular Station—These station modules include docking ports at either end that provide pickup points. For transportation in the shuttle payload bay adapter fixtures will be required to bridge from docking ports to payload bay attach points. Docking ports provide adequate pickup points for orbit-to-orbit transportation. These modules must be protected from aerodynamic loads during Earth launch.

Table 3.4-5. Crew Rotation and Resupply Payloads

PAYLOAD ITEM	DESCRIPTION	MASS*		SIZE (DIAMETER x LENGTH)	
		10^3 KG	10^3 LB	METERS	FEET
1. CTV					
• BASIC VEHICLE	CREW QUARTERS AND SUBSYSTEMS	3.8	8.4	4.4 x 4.3	14.5 x 14
• CONSUMABLES		0.4	0.9	—	—
• CREW & RESERVES	INCLUDES CARRY ON EQUIP & FLUIDS RESERVES	0.7	1.5	—	—
• RETURN SCIENCE		0.2	0.4		
• PROPULSION SYSTEM	EMERGENCY USE ONLY	0.6	1.3	—	—
TOTAL CTV (RETURN)		5.7	12.5	—	—
2. FLUID MODULE					
• LSV USABLE PROPELLANT	LO_2/LH_2 SINGLE STAGE LSV ASSUMED	31.8	70.1	—	—
• OLS FLUIDS		5.9	13	—	—
• SURFACE FLUIDS	LO_2/LH_2 PROPELLANT & CEM	3.0	6.6	—	—
• BOILOFF	CONSUMABLES	0.5	1.1	—	—
TOTAL FM FLUIDS		41.2	90.8	—	—
FM INERT		3.6	7.9	4.4 x 12.2	14.5 x 40
TOTAL FM		44.8	98.7	—	—
3. RESUPPLY MODULE					
• OLS BULK CARGO		3.6	7.9	—	—
• SURFACE BULK		0.9	2	—	—
• TOTAL RM CARGO		4.5	9.9	—	—
• RM INERT		1.8	4.0	4.4 x 5.8	14.5 x 19
TOTAL RM		6.3	13.9	—	—
	DELIVERY TOTAL	56.6	124.7		

*EXCLUDING GROWTH

Table 3.4-6. Consumables

ITEM	DESCRIPTION	MASS	
		KG	LB
CTV	9-DAY MISSION, 4 MEN	400	880
CTV PROPELLANT	USED ONLY IN EMERGENCY	600	1,320
OLS BULK	109 DAYS PLUS 55 DAY CONTINGENCY	3 552	7,830
OLS FLUIDS	SAME AS ABOVE	5 876	12,955
LSV PROPELLANT	LO_2/LH_2 SINGLE-STAGE LSV	31 800	70,110
SORTIE CONSUMABLES	CEM CREW SUPPORT AND MOBILITY SYSTEMS PROPELLANT	3 000	6,600
BOILOFF ALLOWANCE		500	1,100
SORTIE BULK CARGO	FOOD, LiOH, EXPERIMENTS	900	1,985
	TOTAL CONSUMABLES	46 628	102,781

Table 3.4-7. OLS Consumables

	Kg	LBS
FOOD	1,513	3,335
ECLSS	252	555
STATION SPARES	1,624	3,580
STATION CRYOGENICS*		
LH_2	525	1,158
LO_2	1,971	4,345
LN_2	1,894	4,175
EXPERIMENT CRYOGENICS*		
LO_2	—	—
LN_2	57	125
N_2H_4	1,429	3,150
EXPERIMENT SPARES	163	360
	9,460	20,793

*TRANSPORTED IN FLUID MODULE

Table 3.4-8. OLS Mass Summary

ITEM	MODULAR STATION		UNITARY STATION	
	10^3 KG	10^3LB	10^3KG	10^3LB
STATION	100.6	222.0	76.7	169.2
2 LSV'S(TYPICAL	75.6	166.7	75.6	166.7
SORTIE PAYLOAD	13.9	30.6	13.9	30.6
TOTAL ORBITAL MASS	190.1	419.3	166.2	366.5

Unitary Station The large unitary station modules incorporate structural hard points around their aft circumference for adaption to a heavy lift launch vehicle or orbit transfer vehicle payload support structure. Docking ports provide pickup points for module grouping as needed for orbit transfer. The unitary station modules must be provided from aerodynamic loads during Earth launch.

LSV's These are handled like the modular station modules. If launched by the shuttle, they are offloaded to be compatible with shuttle payload capability.

CTV and RM's These are handled like the modular station modules.

Orbit-to-orbit transfer accelerations must be limited to about 5 m/sec^2 (1/2 g) when modules are grouped in assemblies connected by docking structures.

3.4.1.2.3 Orbital Assembly, Maintenance, and Modification

OLS—A range of potential OLS breakdowns exists for transport to the Moon. The degree of orbital assembly in Earth orbit and lunar orbit will depend on the transportation system selection and its delivery capability per trip.

In general, assembly requirements will involve docking of modules or sets of modules together, pressurization, deployment of solar arrays and antennas, checkout, and commissioning. All of these functions will be required to some degree regardless of the choice of transportation.

Once established in lunar orbit, the OLS will be maintained and modified there for its operational life. The impact of these factors on transportation requirements is expected to be incidental compared to the requirements imposed by crew rotation and operational logistics.

Assembly and Operations with LSV—The assembly operation begins with the LSV, and surface exploration payloads all docked at separate locations on the OLS. The LSV will maneuver and dock with one of the mobility vehicle modules.

Separation of the combination is performed and the payload rotated to a position alongside the LSV. It is then moved under positive control down a guideway track to the base of the LSV. This position will minimize the tipping moment during a lunar landing. The same approach is then used on the other payload.

Deployment upon reaching the surface consists of rotating the side of the module that supports the surface down to the surface. The rover can then be moved onto the surface and deployed to its full configuration. To return the rover to OLS, the procedure is reversed. Again, the same approach is used for deployment of the flying vehicles.

3.4.1.2.5 Transfer and Storage

A concept for transferring cargo from the OTV to the OLS is depicted in figure 3.4-4. A utility tug, if available, or one of the LSV's transports cargo from the OTV to the OLS.

Three separate trips are indicated in the concept, although it would be possible for the LSV to transfer all of the cargo to the OLS and then move each module to its correct location.

Once the cargo modules are at the OLS, cargo can be transferred to the appropriate OLS and LSV modules. In some instances, a given module may not require further movement (e.g., RM). In the case of the FM, movement may be required to simplify fluid transfer.

Bulk cargo will primarily be stored in the cryo storage modules, galley module, and crew-quarters module. Cryogenics from the FM will be supplied to the cryo storage modules and electrical power module. Bulk cargo and cryogenics (propellant) will also be transferred to the LSV, and surface payloads while they are docked to the OLS. Propellant and atmosphere cryogenics will be transferred directly from the FM.

3.4.1.3 Transportation Requirements

3.4.1.3.1 Payload Delivery Points

Payload delivery points are summarized in table 3.4-9. Also indicated are representative vehicle utilizations.

3.4.1.3.2 Payload Delivery Options

The modular station is delivered to low Earth orbit one module at a time by the space shuttle; the unitary station is delivered as a single unit by a HLLV. Stations are assembled in low orbit for checkout and then partially disassembled for delivery to the geosynchronous orbit as appropriate to orbit transfer capabilities (see para. 3.4.1.3.2.2 below).

Crew rotation and resupply occur on a combination flight.

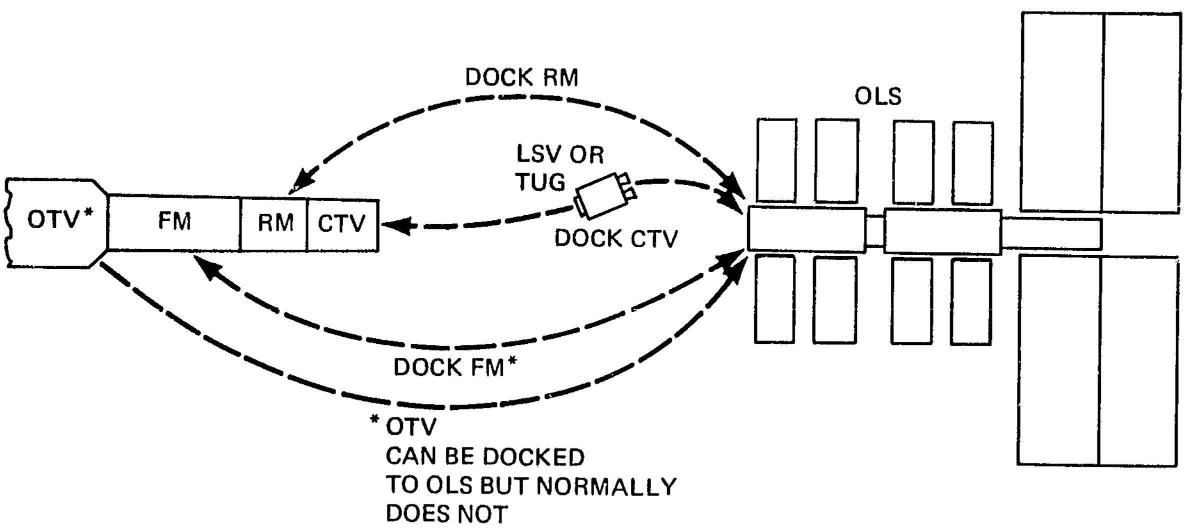


Figure 3.4-4. Logistics Module Docking to OLS

Table 3.4-9. Orbiting Lunar Station Nominal Payload Delivery Points

DELIVERY POINT \ SYSTEM ELEMENT	OLS	ORBIT TRANSFER VEHICLE	OTV TANKER & PRO-PELLANT	CREW	CARGO & EXPERI-MENTS	CREW TRANSFER VEHICLE	LUNAR SORTIE VEHICLE (LSV)
TO EARTH ORBIT	SS OR HL	SS OR HL	SS OR HL	SS	SS	SS OR HL	SS OR HL
EARTH ORBIT TO LUNAR ORBIT	OTV	SELF		OTV	OTV	OTV	OTV
LUNAR ORBIT TO LUNAR SURFACE				LSV	LSV		SELF
LUNAR SURFACE TO LUNAR ORBIT				LSV	LSV		SELF
LUNAR ORBIT TO EARTH ORBIT		SELF		OTV OR LSV OR CTV	OTV	SELF	SELF
EARTH LANDING			SS	SS OR CTV	SS OR CTV	SELF	SS
DISPOSAL	LUNAR CRASH	REUSED; CONTROLLED EARTH ENTRY AT END OF LIFE	REUSED		REUSED OR LEFT ON MOON	REUSED	REUSED

NOTE: SS = SPACE SHUTTLE HL = HEAVY LIFT

Payload values used in sizing the transportation systems included 24% growth allowance on station modules, 20% growth allowance on CEM's and CTV's, and no growth on surface payloads as the latter were considered capabilities rather than requirements. Growth allowances were applied to hardware mass but not to consumables.

3.4.1.3.2.1 Station Delivery

Four buildup or delivery options have been considered. In developing these options, it was assumed that if the OLS modules were to be delivered in less than a complete configuration, a crew would be required in lunar orbit prior to delivery of the remaining modules in order to assist in the docking and checkout.

The first option is the OLS delivered in a single launch as shown in figure 3.4-5. In this case, two suboptions are possible. Option A consists of the entire OLS plus one LSV for a combined weight of 169 600 kg (374,000 lb). Option B is simply delivering the OLS without LSV at a weight of approximately 125 000 kg (276,000 lb).

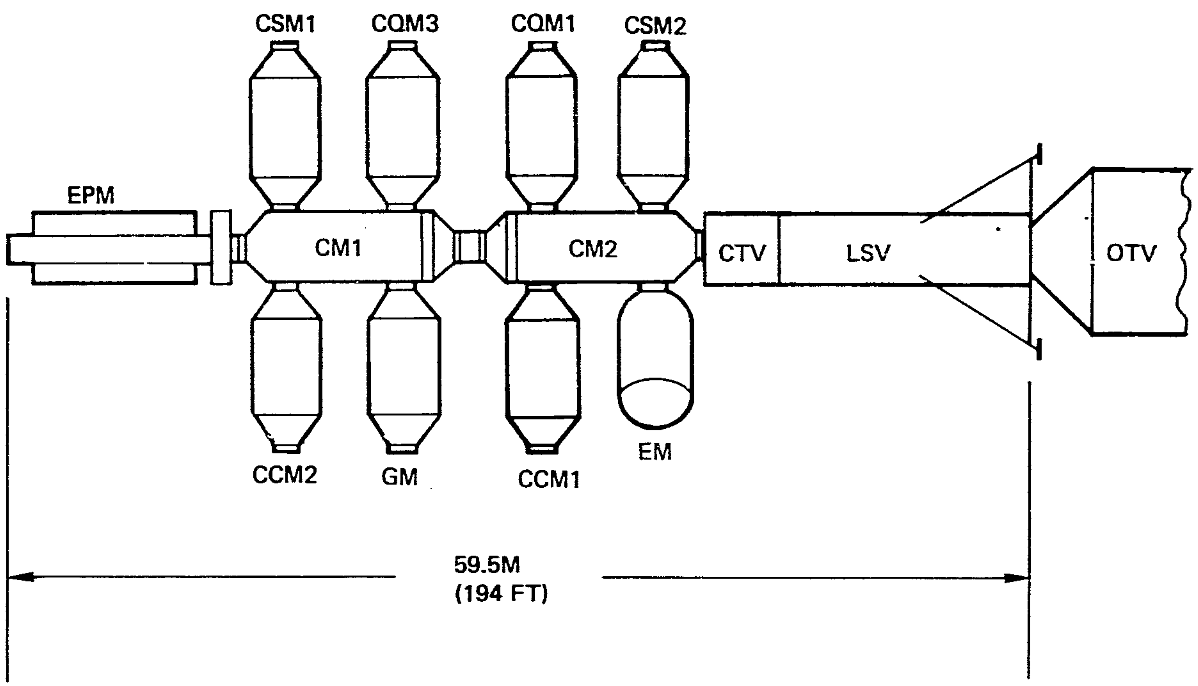
Option 2 consists of a double launch as shown in figure 3.4-6. The first launch delivers nine of the modules; the second launch delivers the final two modules and the crew in an LSV. OTV payload weights are reduced to a maximum of 86 360 kg (190,000 lb); however, individual placement of the last two modules is required. Repositioning the experiment module (XM) to the lower side of the OLS is also required.

The third option, shown in figure 3.4-7, requires three launches to get the basic station to lunar orbit. In this case, the OTV payloads are reduced to a maximum of 58 000 kg (128,000 lb); the length is reduced to a maximum of 27.4m (90 ft). The first launch delivers six of the modules, the second launch a crew in a LSV, and the third launch the remaining five modules. The only repositioning of the modules is the XM to the lower surface.

The fourth concept is that developed by the reference study to accomplish OLS transfer to the moon in a single flight with a high-thrust OTV. This concept is shown in figure 3.4-8 and involves the OLS being transported disassembled; it is assembled in lunar orbit.

Two options have been considered for the delivery of the unitary OLS including one LSV. In the two launch concept illustrated in figure 3.4-9, the basic core and power modules are included in one launch and the LSV, CTV and experiment module in the other launch. The OTV payload masses for the two launches are approximately 79 000 kg (172,000 lb) and 52 600 kg (116,000 lb) including

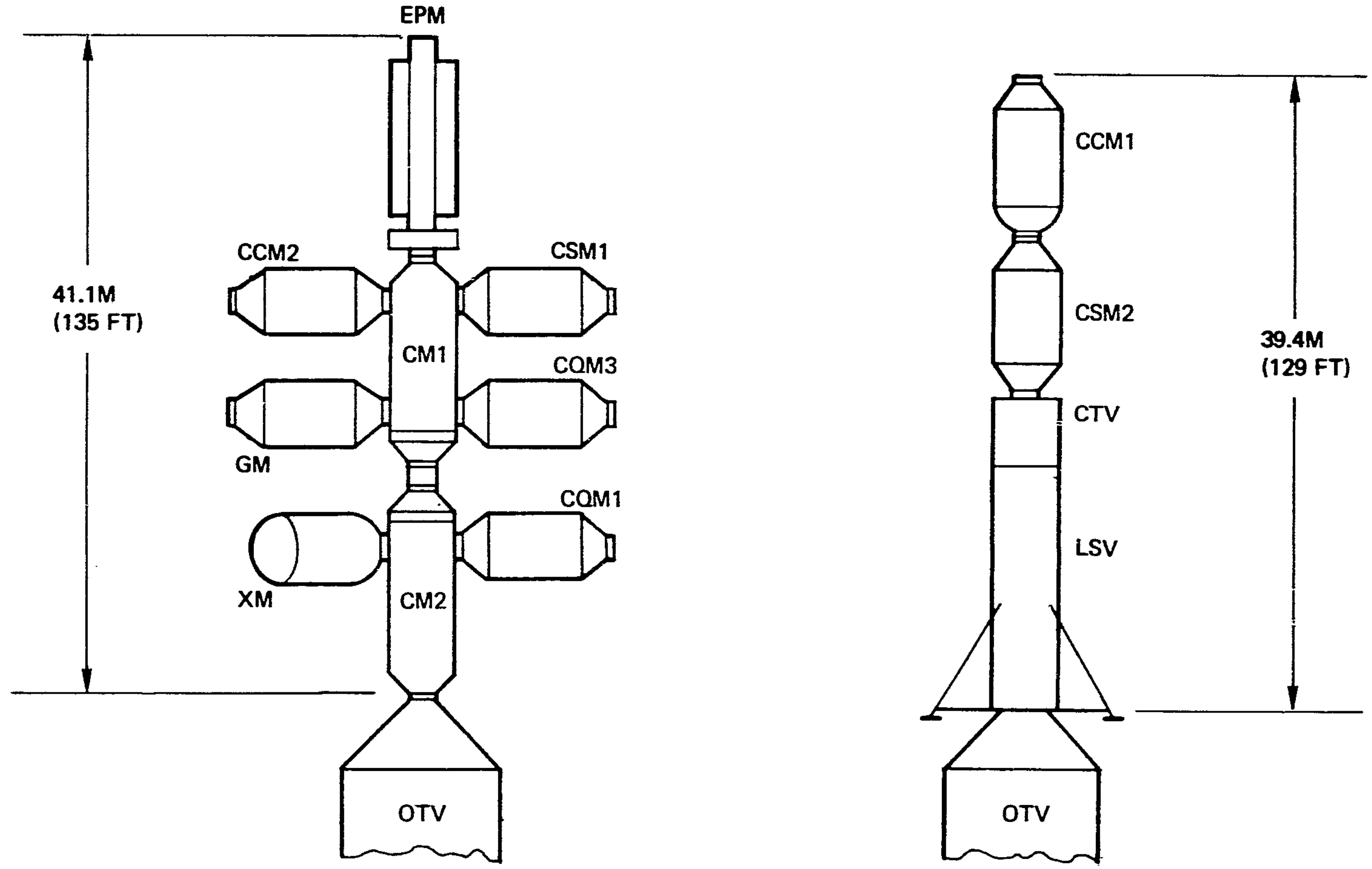
1. ONE FLIGHT



OPTION	OTV PAYLOAD (WITH GROWTH)		
A	WITH LSV	167 730 KG	(369,000 LB)
B	WITHOUT LSV	126 300 KG	(278,000 LB)

Figure 3.4-5. Orbiting Lunar Station Delivery

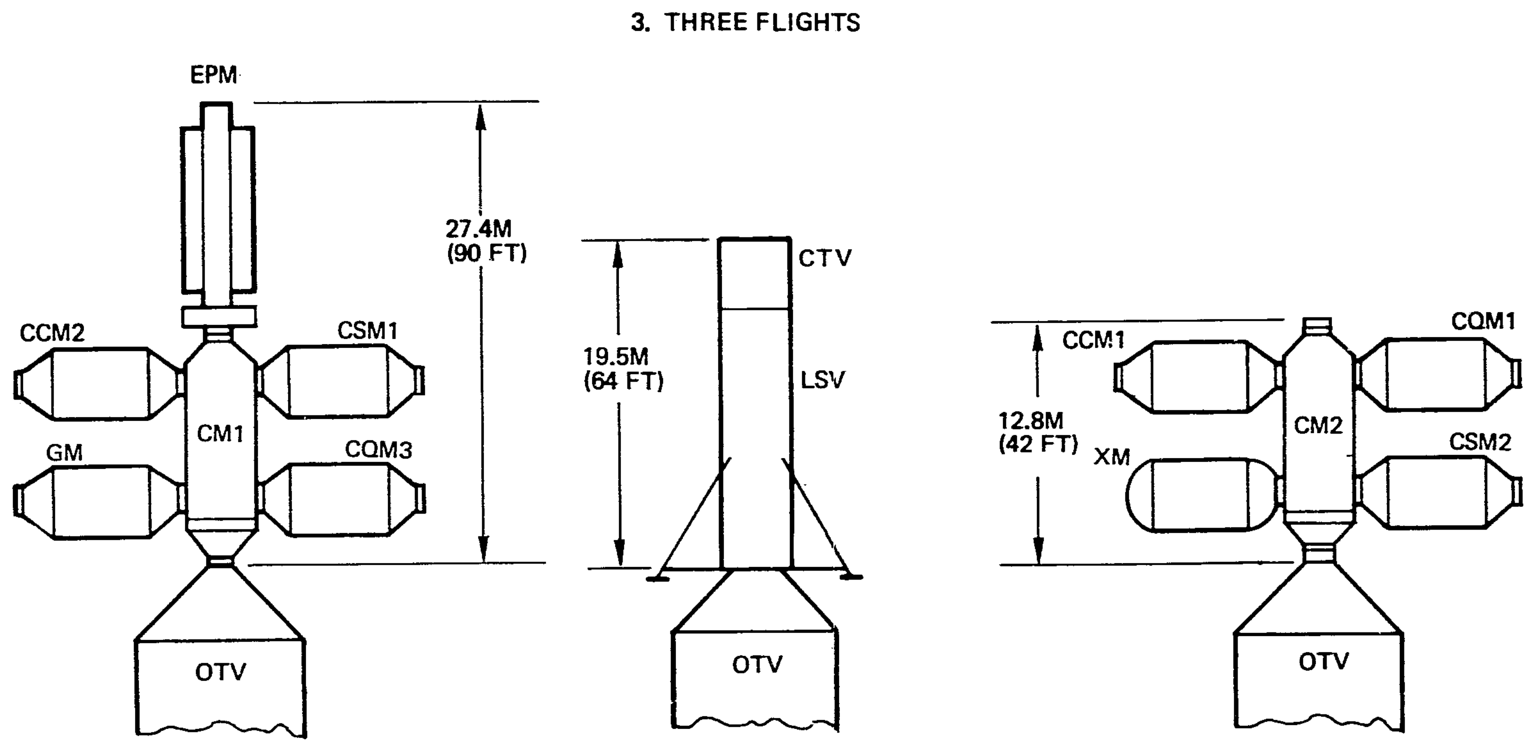
2. TWO FLIGHTS



OTV PAYLOAD (WITH GROWTH)

81,360 KG (179,000 LBS)	86,365 KG (190,000 LBS)

Figure 3.4-6. Orbiting Lunar Station Delivery



OTV PAYLOAD (WITH GROWTH)

58,180 KG (128,000 LB)	49,545 KG (109,000 LBS)	58,180 KG (128,000 LBS)

Figure 3.4-7. Orbiting Lunar Station Delivery

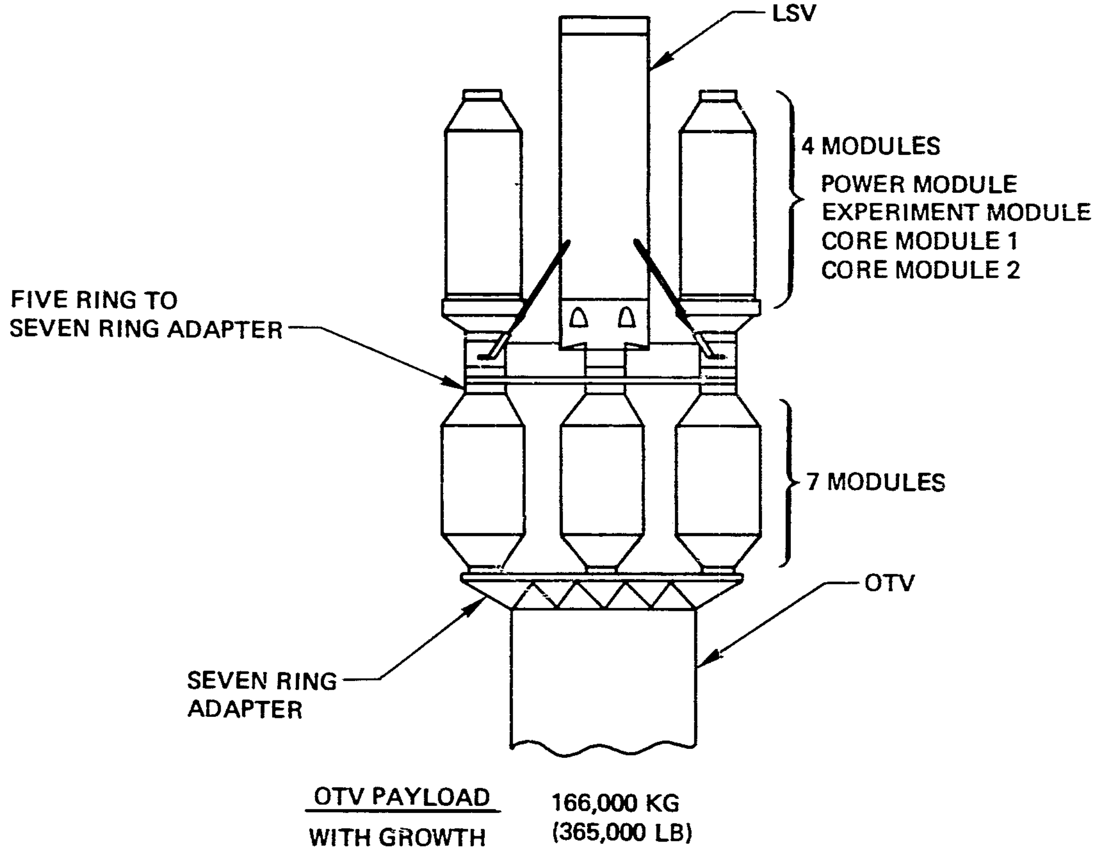
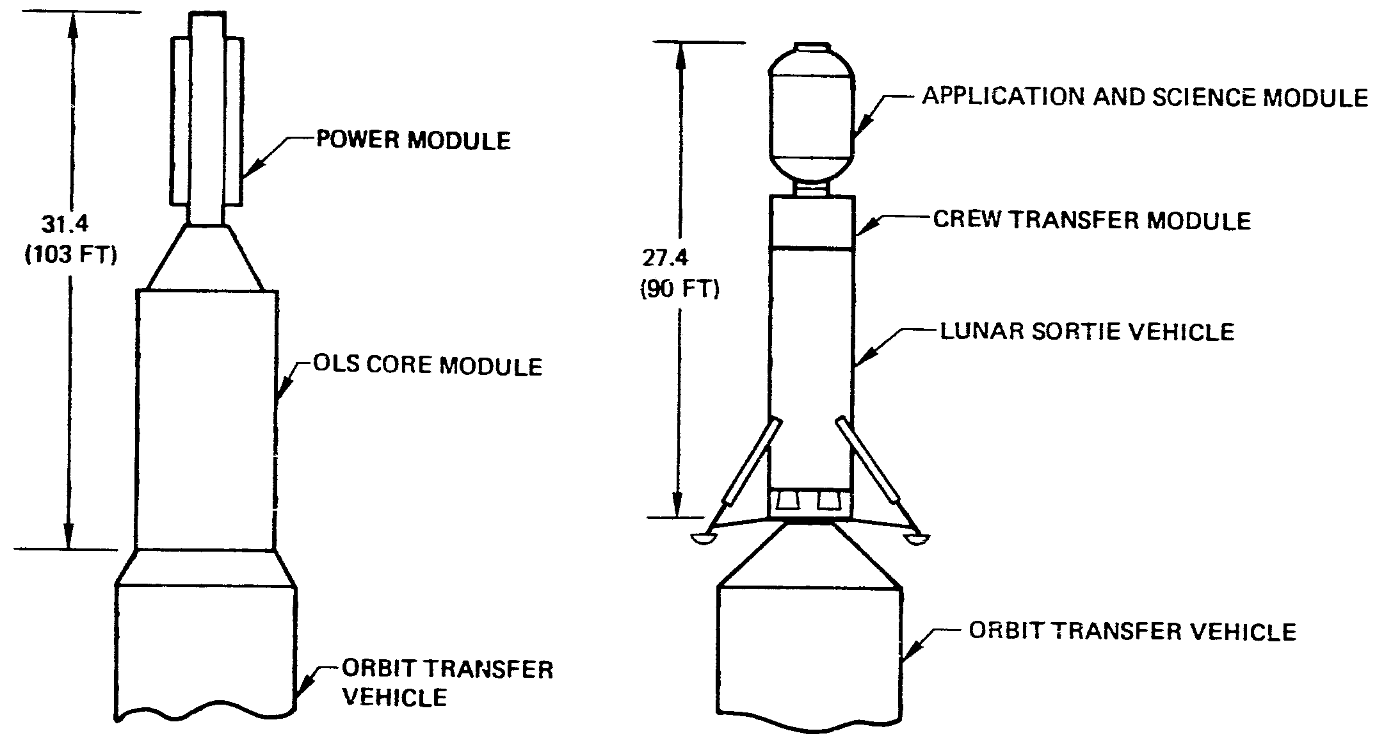


Figure 3.4-8. Disassembled Delivery Concept



DELIVERY OPTION	OTV PAYLOAD CHARACTERISTICS			
	MASS		LENGTH	
	10^3 KG	10^3 LB	M	FT
ONE FLIGHT	131	288	58.8	193
TWO FLIGHTS	78.2; 52.7	172; 116	31.7; 27.4	102; 90

Figure 3.4-9. Unitary OLS Delivery

the expected growth on hardware. A single launch concept results in an OTV payload of approximately 131 000 kg (288,000 lb).

3.4.1.3.2.2 LSV/Crew/Resupply Delivery

Delivery of lunar sortie vehicles, crew and resupply payloads from Earth orbit to lunar orbit requires use of an orbit transfer vehicle (OTV). Delivery concepts are depicted in figure 3.4-10.

Delivery of the second or replacement LSV and 109 days of supplies in the resupply module (RM) results in a payload mass of approximately 62 600 kg (138,000 lbs). In this case, the LSV is fully fueled so no fluid module is required. The purpose of the CTV on this flight is to provide return capability for crew members at the OLS and still retain two LSV's at the OLS. The CTV is located at the forward end of the configuration to simplify separation in the case of an abort.

A typical crew rotation and 109 day resupply configuration results in a payload mass of approximately 58 400 kg (128,750 lbs). The fluid module (FM) in this delivery transports the cryogenics for the OLS and propellant for a fully reusable single state LSV. Should any portion of the LSV propulsion hardware be expendable then the resupply delivery flight must include the appropriate replacement as well as propellant. The CTV will constitute the return payload and with surface samples, film and other will have a mass of 6 100 kg (13,400 lb).

3.4.1.3.2.3 Surface Payload Delivery

As previously described, the LSV surface payloads transported between lunar orbit and the lunar surface include the crew/equipment module to provide quarters and operations center while on the moon, a lunar rover, two lunar flyers and a variety of science instruments.

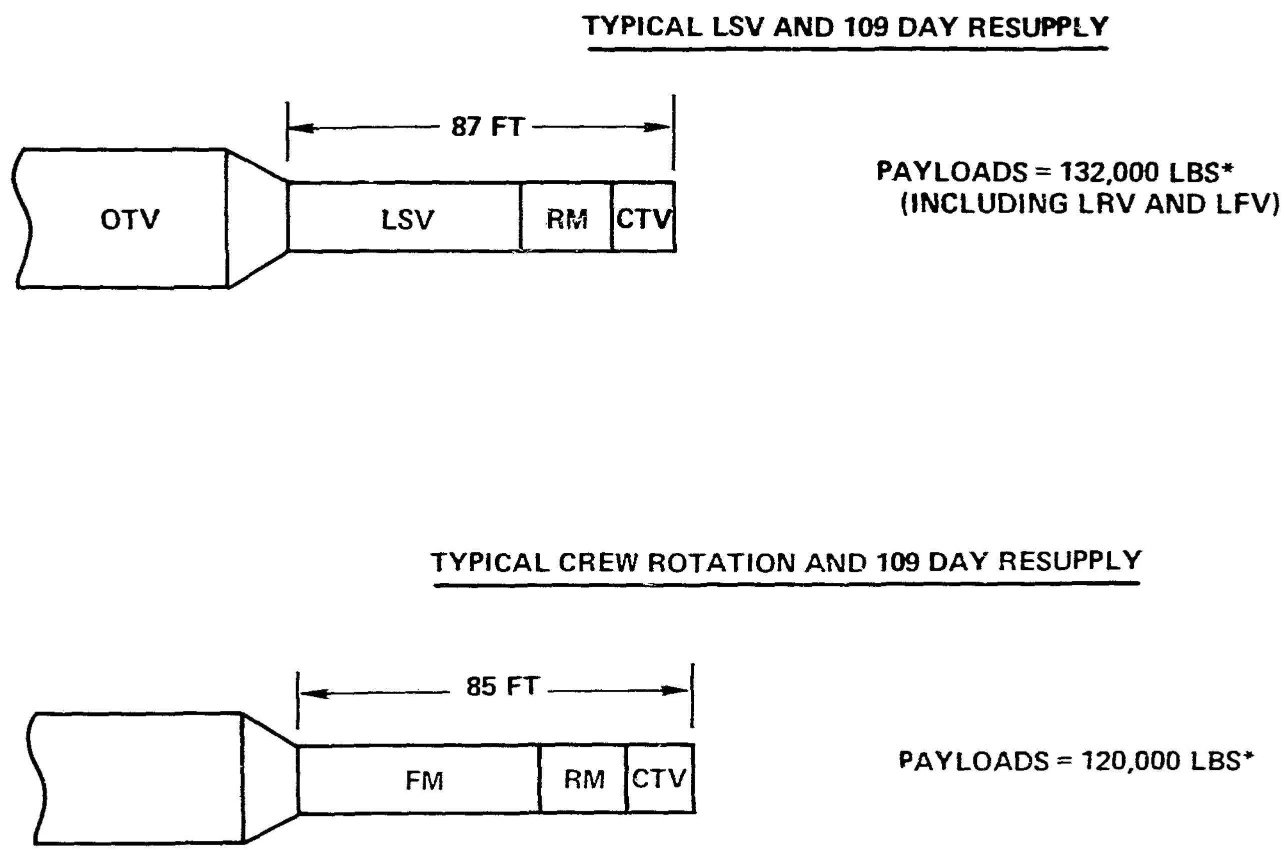
The combined delivery mass of these payloads is 14 900 kg (33,000 lb) with the return payload being 11 500 kg (25,400 lbs).

3.4.1.3.2.4 Operational Constraints

Logistics flights to the Moon and lunar surface sorties from the OLS must observe flight mechanics constraints if system performance requirements are to be held near minimum levels. These constraints arise from the dynamics of the Earth/Moon system and the nature of minimum energy trajectories.

Summary of Constraints—The following is a summary of constraints:

1. For an OLS operation not dependent on Earth-orbit support by an orbiting facility (space station or staging base), transfer opportunities are controlled principally by lunar polar-orbit



*WITH GROWTH

Figure 3.4-10 Crew Rotation/Resupply Delivery

alignment. Translunar opportunities occur every 14 days. Return opportunities occur about 6 days after each translunar opportunity.

2. If the OLS operation is dependent on support by an Earth-orbiting facility, favorable transfer opportunities occur approximately every 55 days, assuming the Earth orbit is synchronized with lunar motion. Wait periods in lunar orbit for favorable return opportunities may range from 7 to 50 days depending on the inclination of the Moon's orbit to the Earth's equator. Wait periods will generally be 20 days or less if moderate trans-Earth injection delta V penalties are accepted.

3. Surface sortie opportunities for minimum energy ascent or descent to any particular site recur at approximately 14-day intervals. Thus, normal surface stay times in multiples of 14 days are indicated. Requirements for abort from the surface to the OLS in an out-of-plane situation can be met through a rescue mode.

4. If a halo orbit is used, transfer opportunities between a particular Earth orbit and the halo orbit occur approximately every 10 days, and transfer between any point on the surface and the halo orbit may be made at any time. Transfers are relatively long about 8 days between Earth orbit and the halo orbit and about 3 days between the halo orbit and the surface.

Abort-Derived Requirements—If the orbit transfer vehicle (OTV) fails to initiate thrust at lunar arrival, the crew module can separate from it, execute a small delta V, and be captured in a lunar orbit to await rescue by a lunar sortie vehicle based at the OLS. If the OTV fails to initiate thrust at Earth arrival, it is not practical to provide an abort delta V great enough to insert the aborted crew module into low Earth orbit; about 3 000 m/sec (9,800 ft/sec) would be needed. A smaller delta V on the order of 400 m/sec (1,300 ft/sec) will place the crew module in a 1-day waiting orbit. The moon's location at the time of abort initiation is such that a substantial phase change and apsidal rotation would be required for a lunar-based LSV to reach the rescue orbit. However, a partially fueled LSV delivered to orbit by the shuttle will be capable of sufficient delta V (about 5 000 m/sec (16,000 ft/sec) to effect the rescue and return to the shuttle.

Two options exist for surface abort: a requirement may be placed on the lunar sortie vehicle to execute a 90° plane change on ascent to cover the abort case when an abort becomes urgently necessary after 7 days on the surface. This is done through a 3-impulse plane change in a 24-hour intermediate orbit. The delta V requirement represents approximately 1 400 m/sec (4,600 ft/sec)

margin above the nominal in-plane ascent delta V budget. This sizes the LSV and results in expenditure of propellant on nominal missions that would otherwise not occur; the propellant required for the 1 400-m/sec (4,600-ft/sec) margin must be delivered through the nominal 4 170-m/sec (13,700-ft/sec) landing and ascent delta V, even if it is not used. The difference between normal and abort ascent payload reduces, but does not eliminate, this penalty.

The alternative is that, in the event of an urgent out-of-plane abort, the LSV on the surface aborts to an in-plane orbit, achieves whatever plane change it can with remaining propellant, and the crew is rescued by the LSV stationed at the OLS. This reserve LSV is required anyway to provide for rescue off the surface if necessary. It has enough delta V to do the 90° plane change and back. Three-impulse plane changes are used (figure 3.4-11). If a maximum delta V of 1 900 m/sec (6,200 ft/sec) is allowed for the rescue maneuver (one way), the minimum size intermediate orbit has a semimajor axis of 2 550 km (1,380 nmi) at about 3.1 hours. A phasing allowance of up to 2 hours gives a maximum intermediate orbit period of 5.1 hours. As noted in figure 3.4-11, the maximum time to reach the aborted vehicle is approximately 9 hours. This alternative was selected for transportation sizing.

3.4.1.4 Mission/Transportation Modes and Operations

3.4.1.4.1 Transportation Options

Transportation modes for the OLS mission include various staging techniques and propulsion technologies.

The principal OTV transportation candidates are as follows:

LO_2/LH_2 single stage, reusable.

LO_2/LH_2 1-1/2 stage system with a reusable main stage and expendable drop tanks.

LO_2/LH_2 common stage system consisting of two equal size systems, both reusable.

LO_2/MMH common stage system consisting of two equal sizes, both reusable.

The principal LTV transportation candidates are as follows:

- LO_2/LH_2 single stage
- LO_2/LH_2 1-1/2 stage
- LO_2/LH_2 two stage
- LO_2/MMH single stage

MISSION EVENT	TIME REQ'D
ABORT TO ORBIT	1
OLS REACH LINE OF NODES	1
TRANSFER ORBIT	5
RENDEZVOUS	2
TOTAL	9

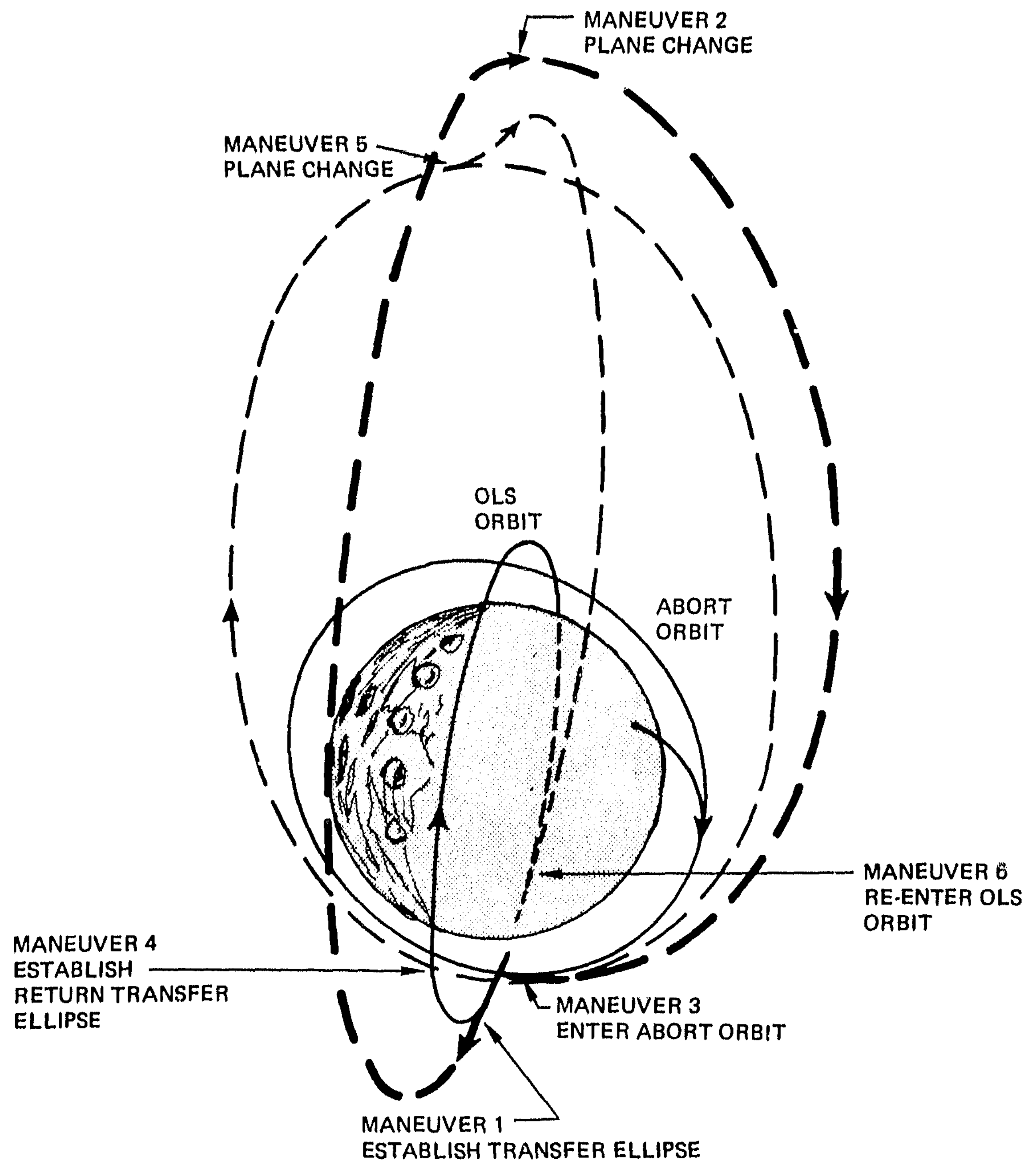


Figure 3.4-11. Out-of-Plane Orbital Rescue

The principal Earth launch vehicle candidates include:

- Space Shuttle (SS) reusable
- Heavy Lift Launch Vehicle (HLLV) expendable

3.4.1.4.2 Representative Transportation Mode and System

3.4.1.4.2.1 Sequence Description

Typical sequences and operations associated with the OLS mission are illustrated in figure 3.4-12. Transportation systems employed include a single stage OTV, single stage LSV and both the space shuttle and HLLV for Earth launches. The principal transportation feature associated with mission are as follows:

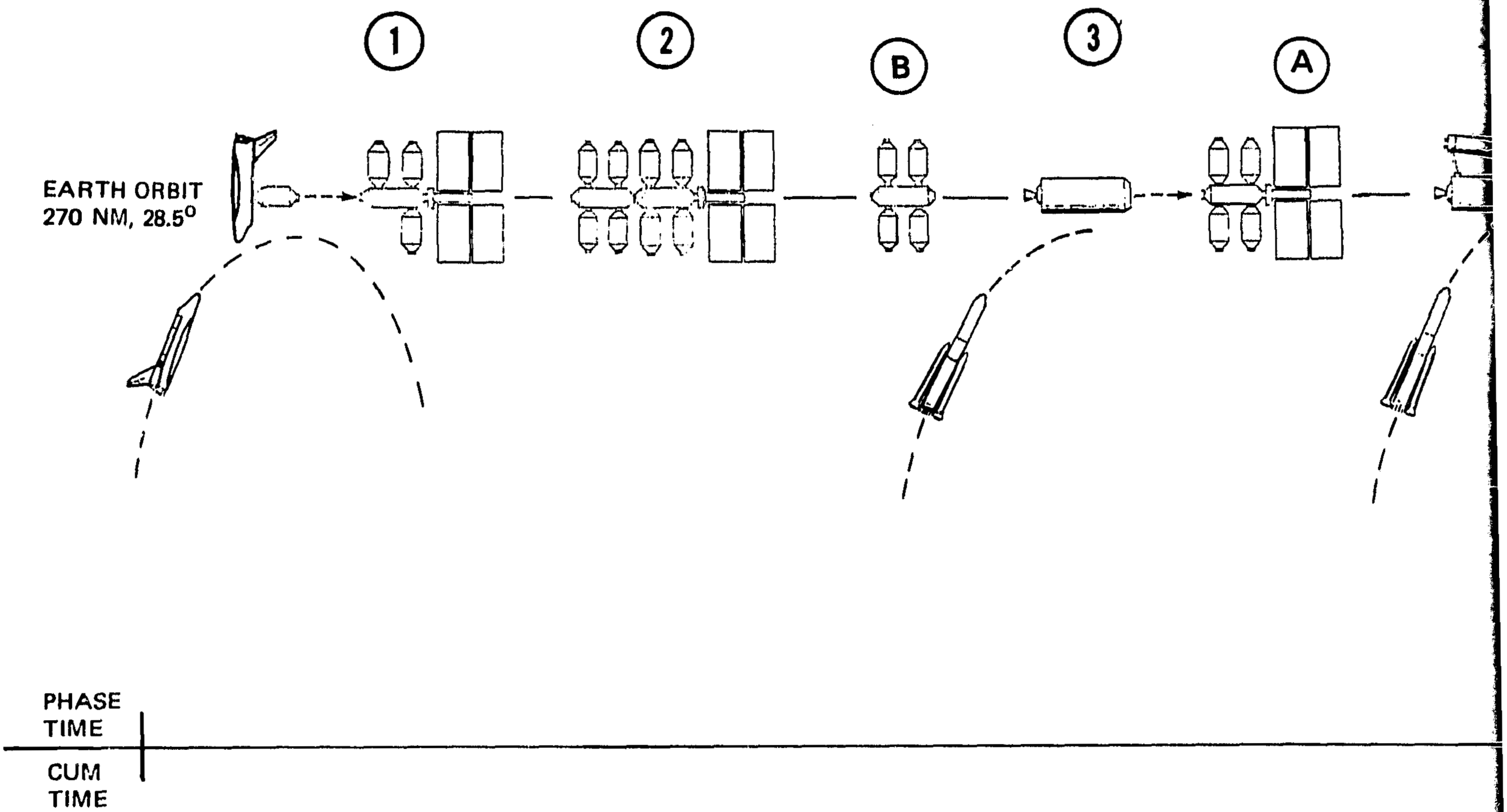
- Station modules are delivered to Earth orbit by the SS, assembled and checked out.
- The OTV is delivered to LEO by a HLLV, docks with one of the station clusters and is subsequently fueled by a tanker also launched by an HLLV.
- The OTV delivers the station cluster to lunar orbit and returns to Earth orbit.
- The first LSV is delivered to Earth orbit by a SS and docks with the OTV. Tankers are again launched with HLLV and refuel the OTV which then delivers the LSV and first crew to the station in lunar orbit.
- Upon return to Earth orbit the OTV docks with the second station cluster, is refueled and delivers the cluster to lunar orbit.
- Delivery of the second LSV to the OLS is performed in the same manner as for the first LSV.
- Surface sorties will be performed by the LSV.
- Crew rotation/resupply flights will use an OTV to deliver the payloads to the near vicinity of the OLS at which time a LSV will receive the payloads and take them to the station.
- Crews will be transported back to Earth orbit in the CTV by the OTV where they will be retrieved by the SS.

A flight profile history including elapsed time, ΔV, and weight history is presented in table 3.4-10.

3.4.1.4.2.2 Transportation Sizing

The representative LSV was a single-stage LO_2/LH_2 vehicle. Parametric performance for this option is shown on figure 3.4-13. The LSV sizing is a principal factor in OTV sizing. A single-stage LO_2/LH_2 OTV was used for the representative system.

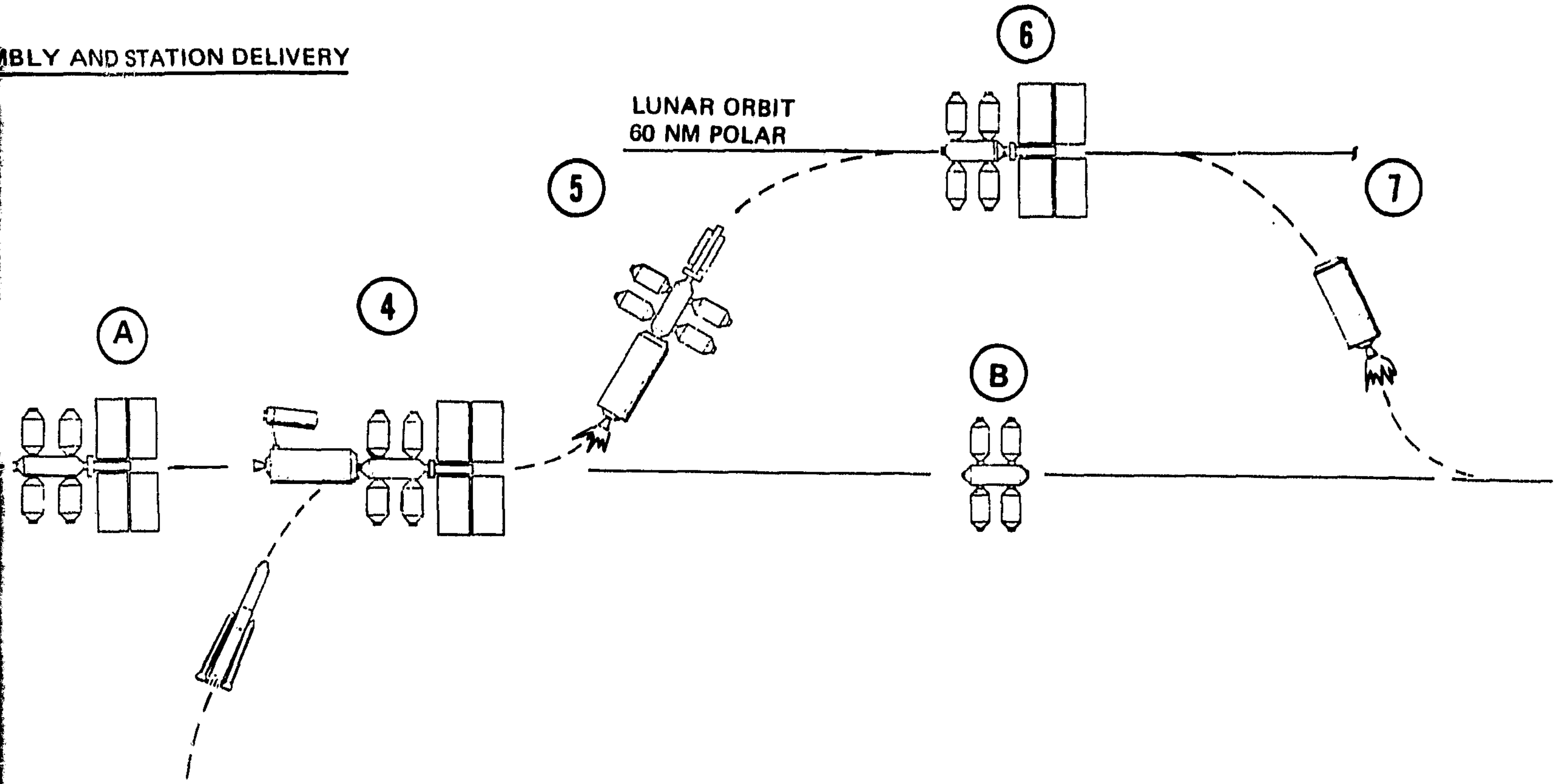
ASSEMBLY AND STATION DELIV



1	2	3	
• DELIVER 10 STATION MODULES, 1 EXPERIMENT MODULE AND CHECKOUT CREW WITH SS • 11 SS FLIGHTS	• CHECKOUT ASSEMBLED STATION	• SEPARATE STATION INTO TWO CLUSTERS (A AND B) • DELIVER OTV WITH HLV • DOCK OTV WITH CLUSTER A	• DE WIT • FUE • DIS

FOLDOUT FRAME

IBLY AND STATION DELIVERY



3	4	5	6	7
TATION INTO RS (A AND B) V WITH HLV ITH CLUSTER(A)	• DELIVER OTV TANKERS WITH HLV (2 FLIGHTS) • FUEL OTV • DISPOSE OF TANKERS	• STOW SOLAR ARRAY • DELIVER CLUSTER(A) TO LUNAR ORBIT WITH OTV	• DEPLOY SOLAR ARRAY • ACTIVATE SUBSYSTEMS	• RETURN OTV TO EARTH ORBIT

Figure 3.4-12 OLS Mission Transportation Sequence (Sheet 1)

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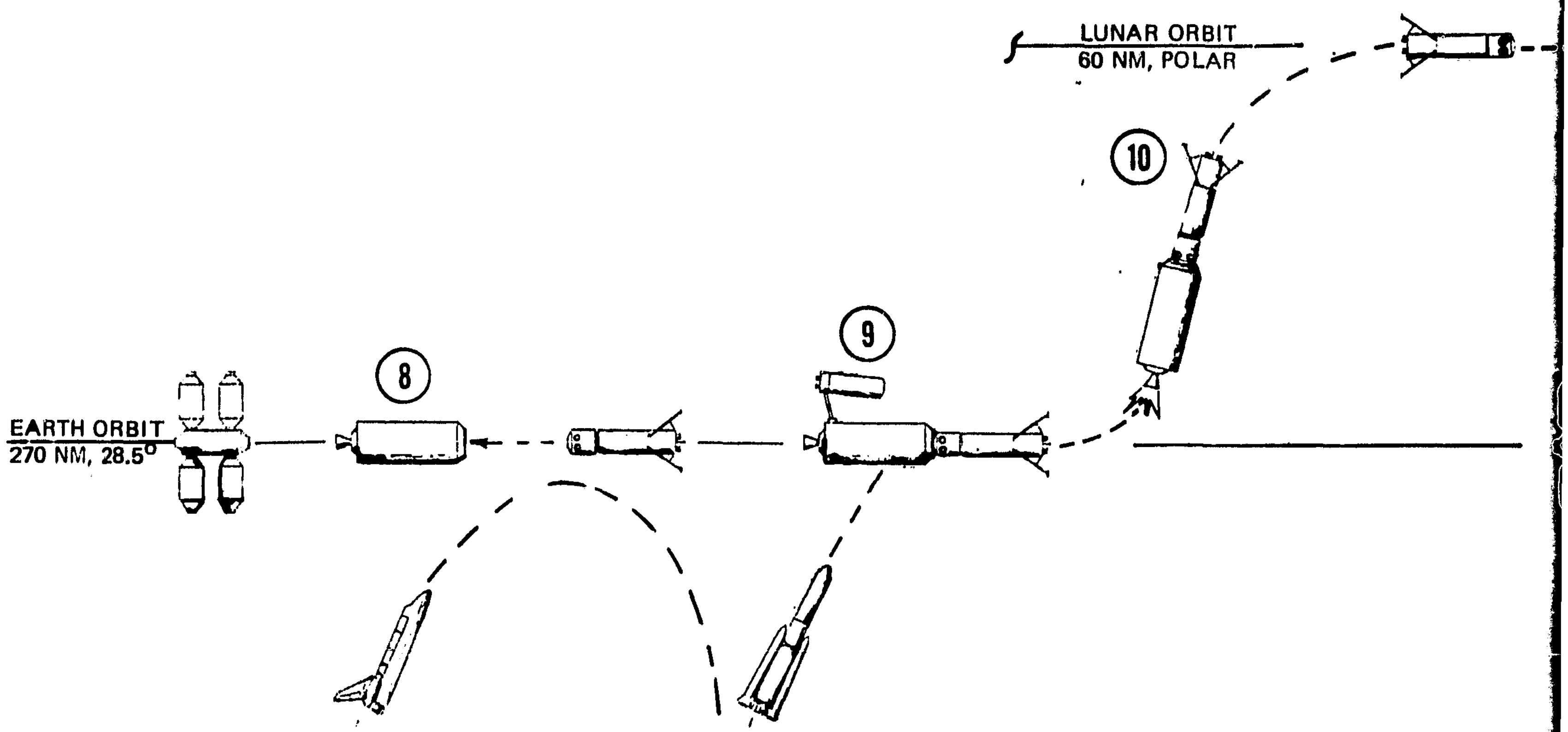
1ST LSV DELIVERY



PHASE TIME

CUM TIME

8	9	10	
• DELIVER LSV AND CREW WITH SS • DOCK LSV WITH OTV	• DELIVER OTV TANKERS WITH HLV (2 LAUNCHES) • FUEL OTV AND LSV • DISPOSE OF TANKERS	• DELIVER LSV TO LUNAR ORBIT WITH OTV	• SEPARATE L • DOCK LSV T CLUSTER (A)

FOLDOUT FRAME

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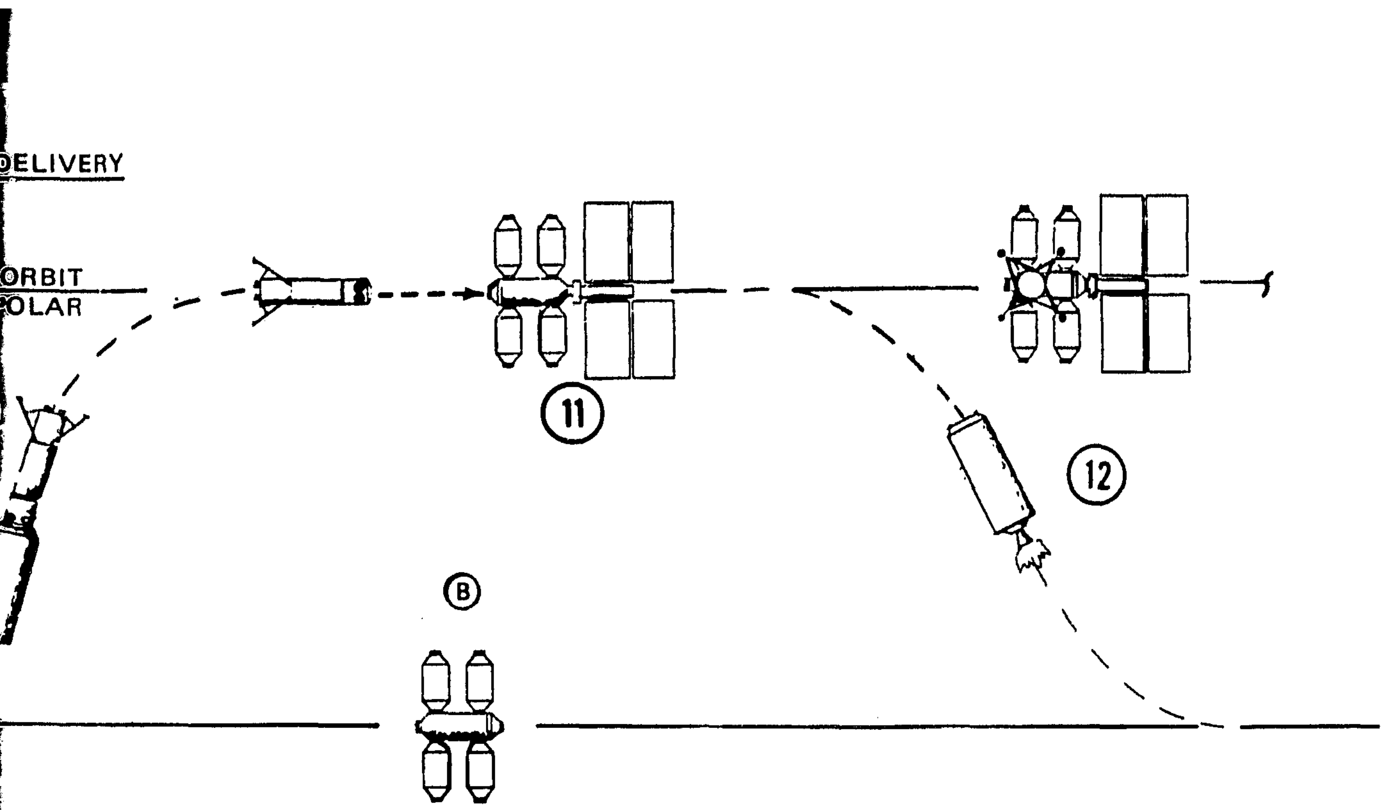
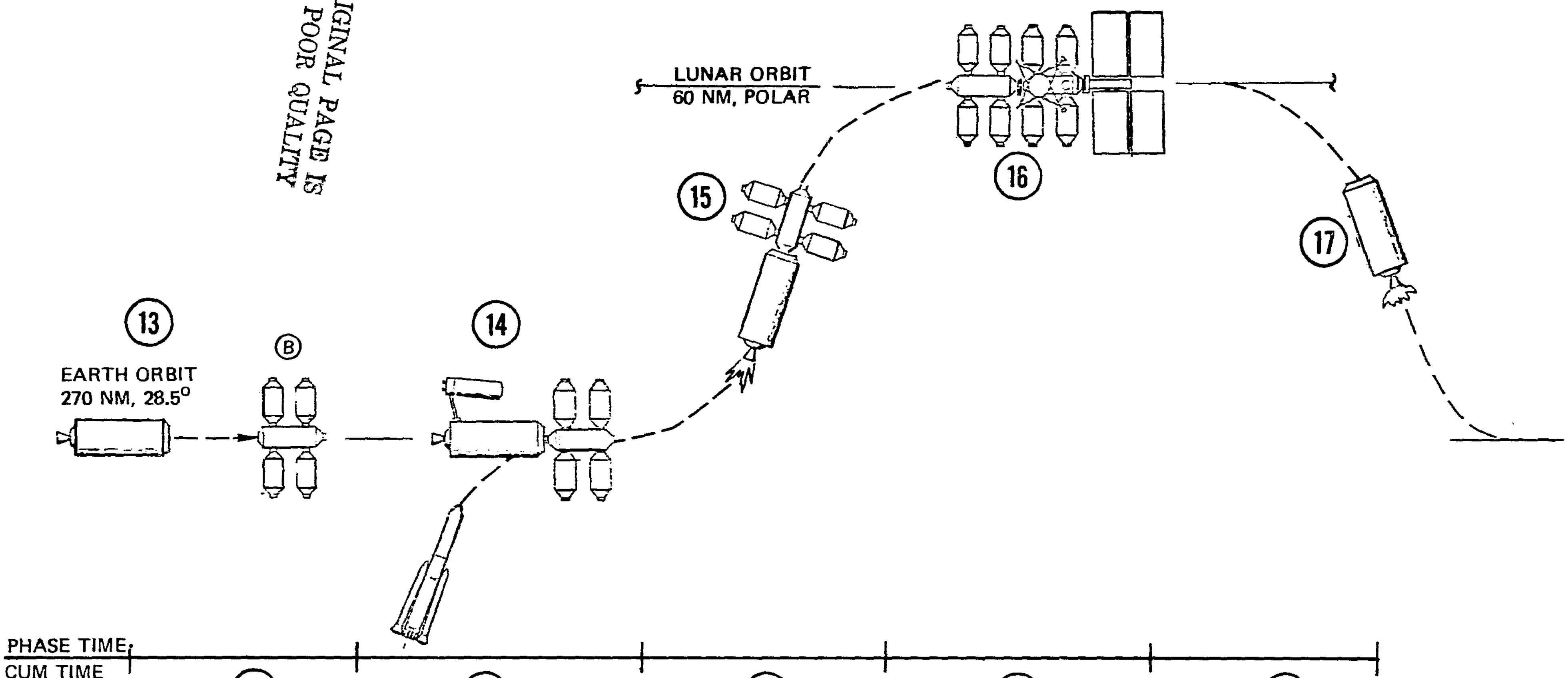


10	11	12
VER LSV INAR ORBIT OTV	• SEPARATE LSV FROM OTV • DOCK LSV TO STATION CLUSTER (A)	• RETURN OTV TO EARTH ORBIT

Figure 3.4-12 OLS Mission Transportation Sequence (Sheet 2)

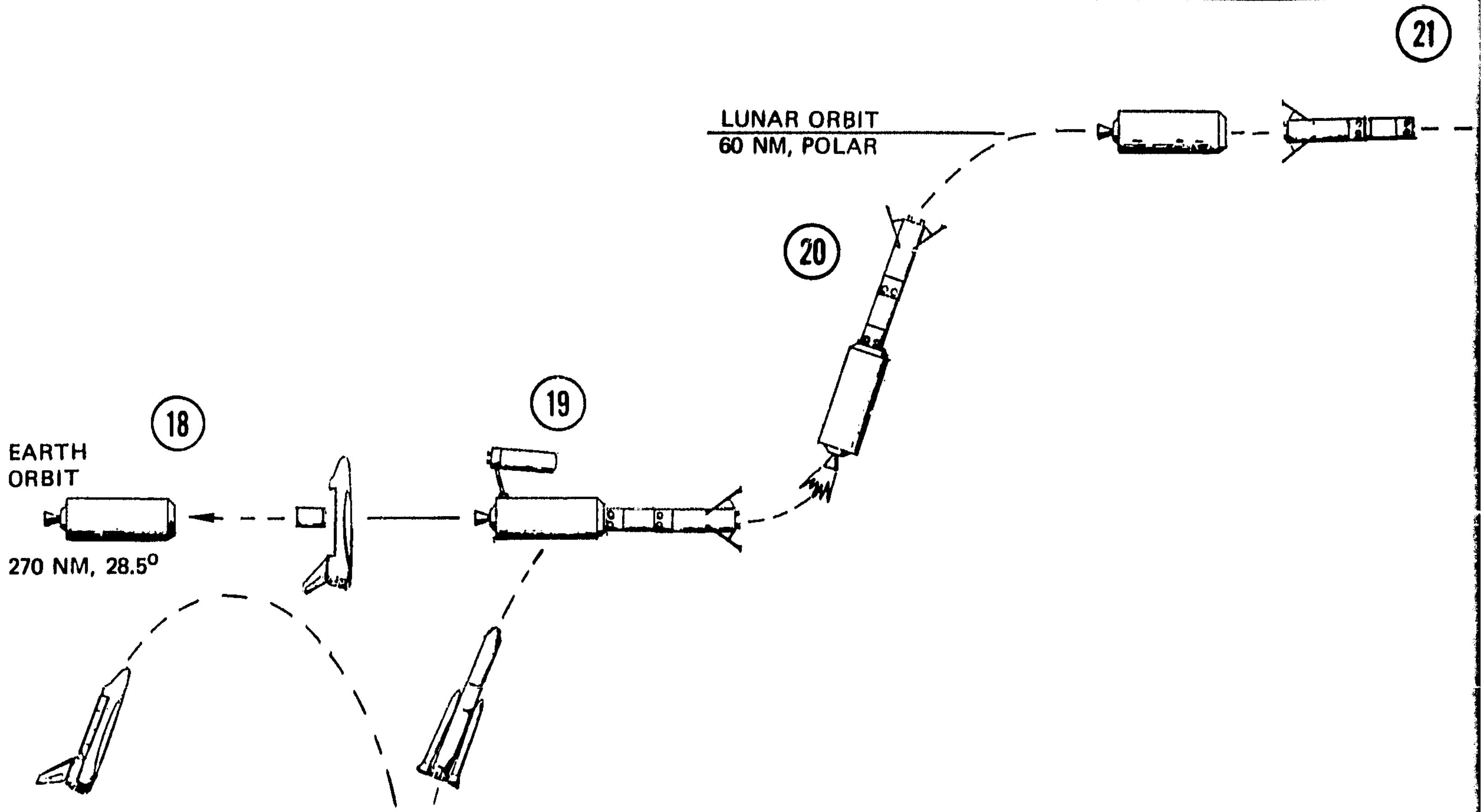
STATION DELIVERY (CONTINUED)



13	14	15	16	17
• DOCK OTV WITH STATION CLUSTER Ⓑ	• DELIVER OTV TANKERS WITH HLV (2 LAUNCHES) • FUEL OTV • DISPOSE OF TANKERS	• DELIVER CLUSTER Ⓑ TO LUNAR • ORBIT WITH OTV	• DOCK CLUSTER Ⓑ TO CLUSTER Ⓐ TO FORM COMPLETE OLS • CHECKOUT ASSEMBLED STATION • MAY USE LSV TO MANEUVER AND DOCK CLUSTER Ⓑ	• RETURN OTV TO EARTH ORBIT

Figure 3.4-12 OLS Mission Transportation Sequence (Sheet 3)

SECOND LSV AND RESUPPLY DELIV



PHASE TIME

CUM TIME

18	19	20	
• DELIVER RESUPPLY MODULE, CREW TRANSFER MODULE AND SECOND LUNAR SORTIE VEHICLE AND OPERATIONAL CREW WITH SS (2 FLIGHTS) • DOCK WITH OTV	• DELIVER TANKERS WITH HLV (2 LAUNCHES) • FUEL OTV AND LSV • DISPOSE OF TANKERS	• DELIVER LSV/CTV/RM TO LUNAR ORBIT WITH OTV	• SEPAR AND D • CHEC USING

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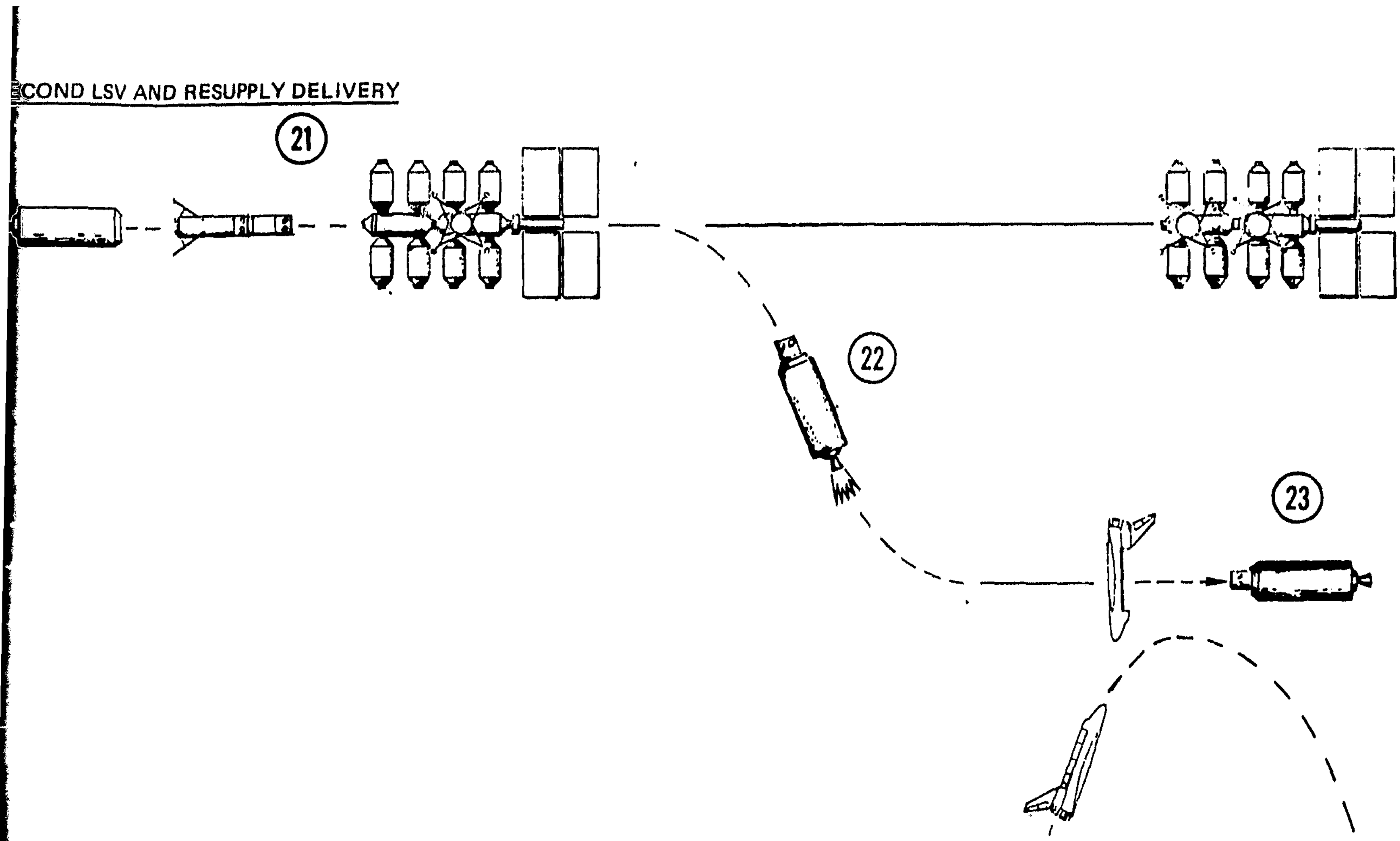
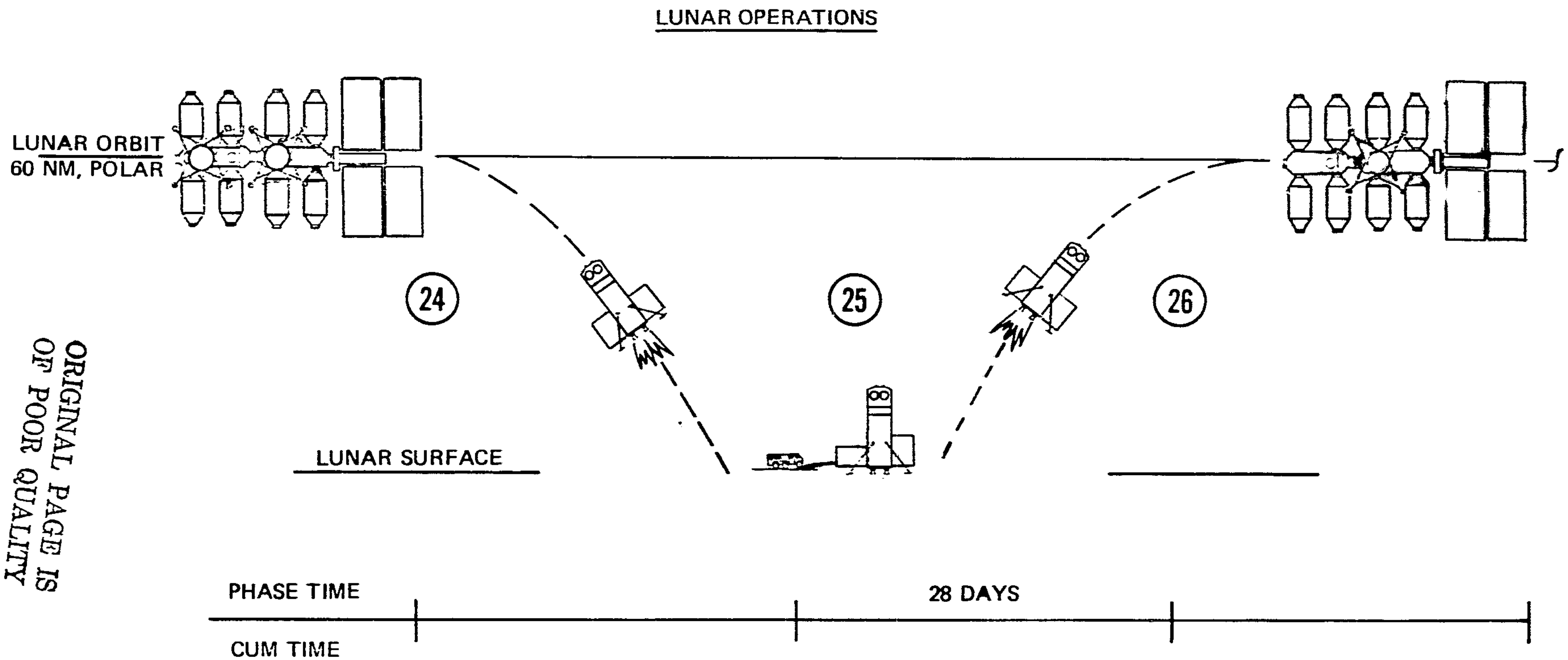


20	21	22	23
LIVER LSV/CTV/RM LUNAR ORBIT H OTV	• SEPARATE LSV/CTV/RM FROM OTV AND DOCK WITH STATION • CHECKOUT CREW RETURN TO OTV USING CTV	• RETURN CTV (CREW) TO EARTH ORBIT USING OTV	• RETRIEVE CTV CREW AND RETURN TO EARTH USING SS

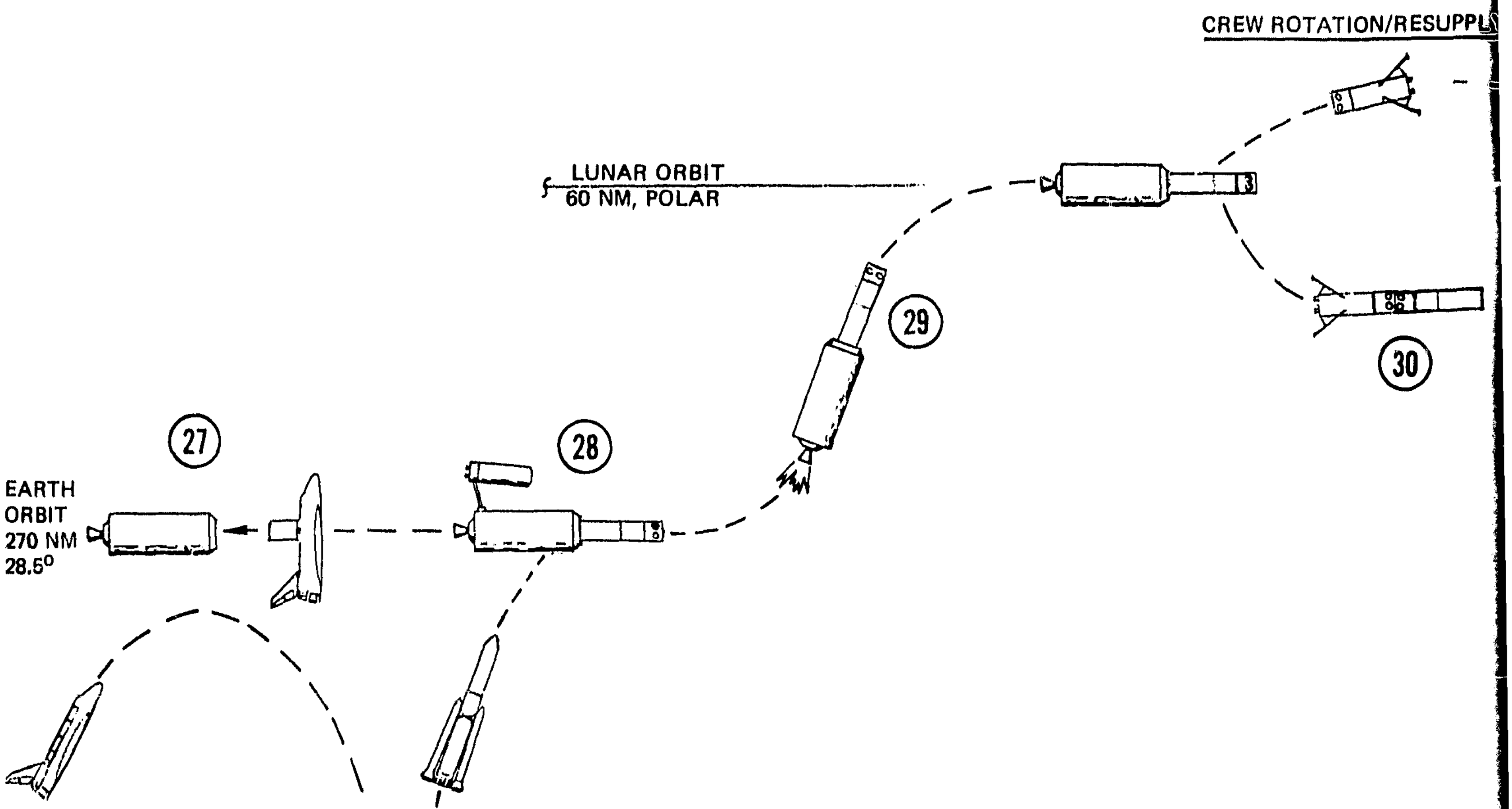
Figure 3.4-12 OLS Mission Transportation Sequence (Sheet 4)

FOLDOUT FRAME 2



24	25	26
• LSV DESCENT AND LANDING WITH 4 CREWMEN	• CONDUCT SURFACE EXPLORATION AND SCIENTIFIC INVESTIGATION USING ROVER AND FLYERS • OLS CONDUCTS ORBITAL SCIENCE AND SUPPORTS SURFACE PARTY	• LSV ASCENT AND RENDEZVOUS WITH OLS • ROVER AND FLYER PAYLOADS RETURNED

Figure 3.4-12 OLS Mission Transportation Sequence (Sheet 5)



PHASE TIME / CUM TIME	27	28	29	
	• DELIVER RESUPPLY MODULE (RM), FLUID MODULE(FM) AND CTV (4 NEW CREWMEN) WITH SS (2 FLIGHTS) • DOCK WITH OTV	• DELIVER TANKERS WITH HLV (2 LAUNCHES) • FUEL OTV AND FM • DISPOSE OF TANKERS	• DELIVER CTV/RM/FM TO LUNAR ORBIT USING OTV	• SEPARATE LSV FR TO CTV/RM/FM • RETURN CTV/RM/F LOCATION USING • REPLENISH LSV PR • RETURN ONE HAL (OLD RM AND DEP USING CTV

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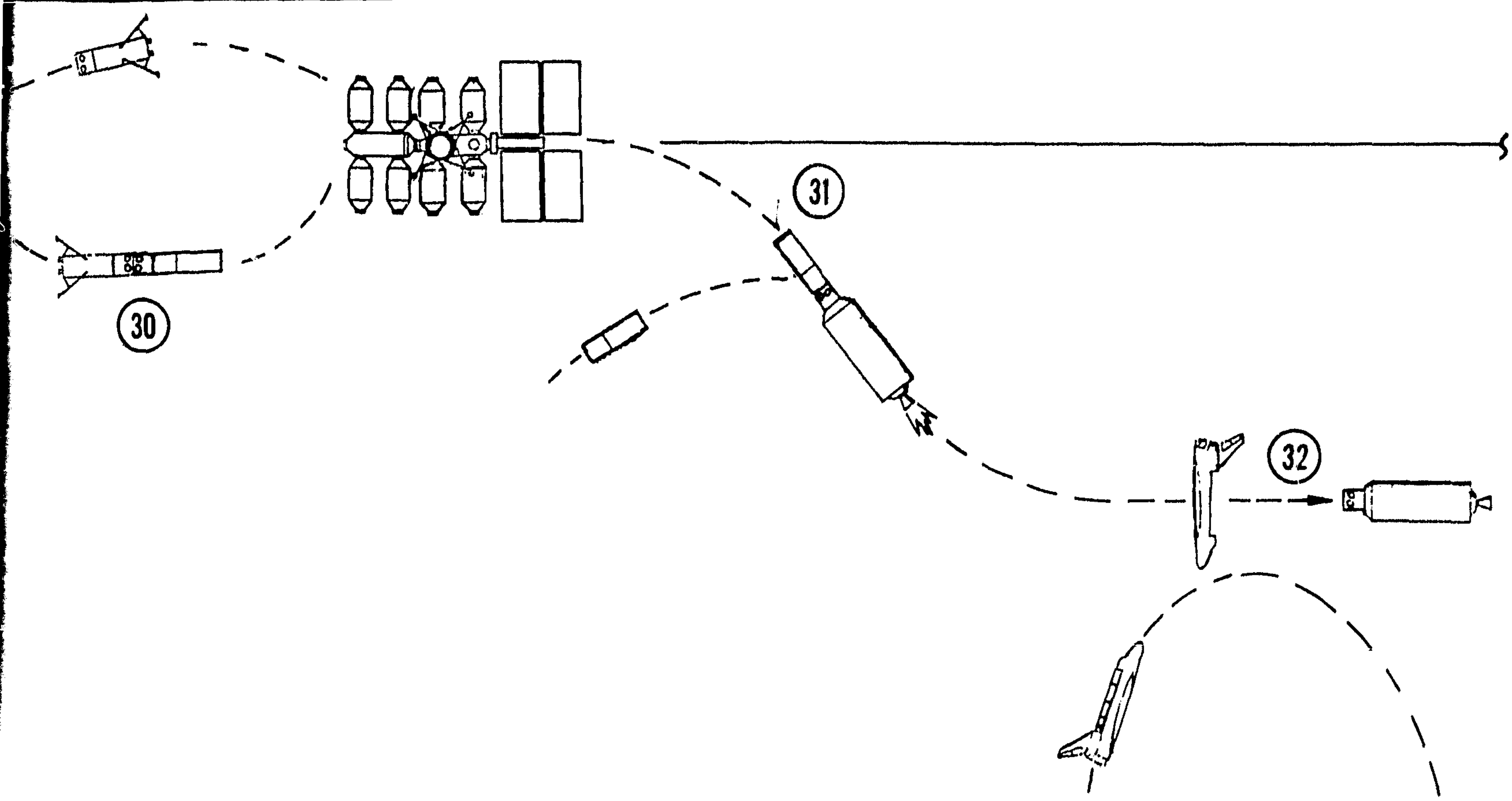
ROTATION/RESUPPLY OPERATIONS



30	31	32
• SEPARATE LSV FROM STATION AND DOCK TO CTV/RM/FM • RETURN CTV/RM/FM TO PROPER STATION LOCATION USING LSV • REPLENISH LSV PROPULSION USING FM • RETURN ONE HALF (4) OF OLS CREW, (OLD RM AND DEPLETED FM), TO OTV USING CTV	• RETURN CTV (CREW) TO EARTH ORBIT USING OTV • JETTISON RM AND FM PRIOR TO TEI	• RETRIEVE CTV (CREW) AND RETURN TO EARTH USING SS • REPEAT STEPS 27 THRU 32 EVERY 109 DAYS

Figure 3.4-12 OLS Mission Transportation Sequence (Sheet 6)

Table 3.4-10 OLS Mission History — Single Stage LO_2/LH_2 OTV

Event	Elapsed Time	ΔV		Mass Remaining	
	Hr	MPS	FPS	KG	LB
Initial mass	0			319,000	703,300
Translunar injection	0.5	3,345	10,974	149,500	329,600
Lunar orbit insertion	90.5	968	3,175	119,900	264,300
Separate payload 58,000 kg (127,900 lb)	100			61,900	136,500
Return payload 7,000 kg (15,400 lb)	440			68,900	151,900
Transearth injection	450	968	3,175	55,400	122,100
Earth orbit insertion	565	3,198	10,491	27,000	59,500
(OTV inert)				(20,000)	(44,100)
(Payload)				(7,000)	(15,400)

Table 3.4-10. (Continued) OLS Mission History — Single Stage LO_2/LH_2 LSV

Event	Elapsed Time	ΔV		Mass Remaining	
	Hr	MPS	FPS	KG	LB
Initial mass				53,700	118,400
Descent	2	2,194	7,198	32,400	71,500
Offload landed payload 4,500 kg (10,000 lb)	10			27,900	61,500
Ascent	676	2,020	6,627	17,400	37,300
(LSV inert)				(6,000)	(13,200)
(Ascent payload)				(11,400)	(25,100)

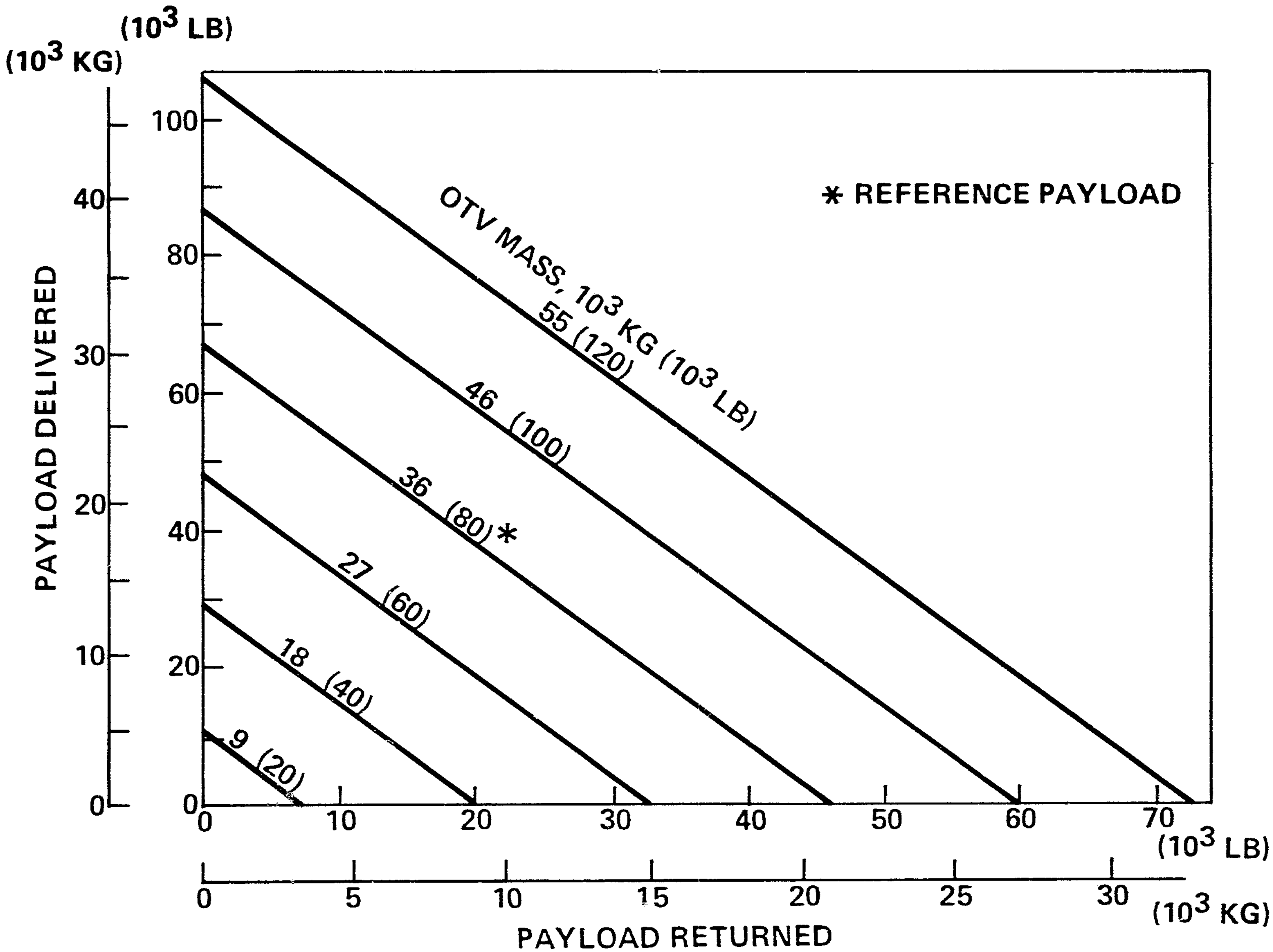


Figure 3.4-13. Single Stage LO_2/LH_2 LSV Capability for OLS

As indicated, on figure 3.4-14, a wide range of OTV sizes is possible to accomplish the delivery of the station and crew rotation/resupply equipment. However, since there will be many crew rotation/resupply flights (3 per year) during the course of the program as opposed to only several deliveries for the station, OTV size selection is based on the crew rotation/resupply requirement. Delivery of the modular station therefore should be accomplished in three flights (one flight includes one LSV).

The unitary station concepts could be delivered in two flights.

3.4.1.4.2.3 **Operational Factors**

Mission Profiles and Rendezvous Techniques—Nominal mission profiles require transfers between low Earth orbits and lunar polar orbits. The profiles are governed by flight mechanics constraints as described in paragraph 3.4.1.3.2.4. A typical Earth-moon profile employs a 90-hour outbound transit, 15 days' stay in lunar orbit, a 110-hour inbound transit, and about 5^{o} plane change entering and leaving lunar orbit.

Target orbits entering lunar and Earth orbits will be about 18 km (10 nmi) below the operational orbits at 528 km (285 nmi) and 111 km (60 nmi) at Earth and moon, respectively. After orbit insertion, target orbits will be trimmed coplanar and a conventional concentric rendezvous sequence used.

Crew Involvement and Timelines—Crew involvements are summarized in table 3.4-11. Crew timelines as such do not impose any identified transportation requirements.

Control Functions and Operational Requirements—Precision targeting is required for Earth orbit insertion, lunar orbit insertion, and lunar landing. These control functions should be autonomous with network support only as a backup.

The following operational requirements were identified:

1. The propellant tanker for the orbit transfer vehicle (OTV) must rendezvous with the OTV and make sufficient physical contact to accomplish propellant transfer.

2. The orbit transfer vehicle must be capable of rendezvous and docking with the OLS and with a support facility in Earth orbit if one is used.

3. The OLS must provide docking facilities for the OTV for two LSV's and for six to eight cargo and sortie payload modules.

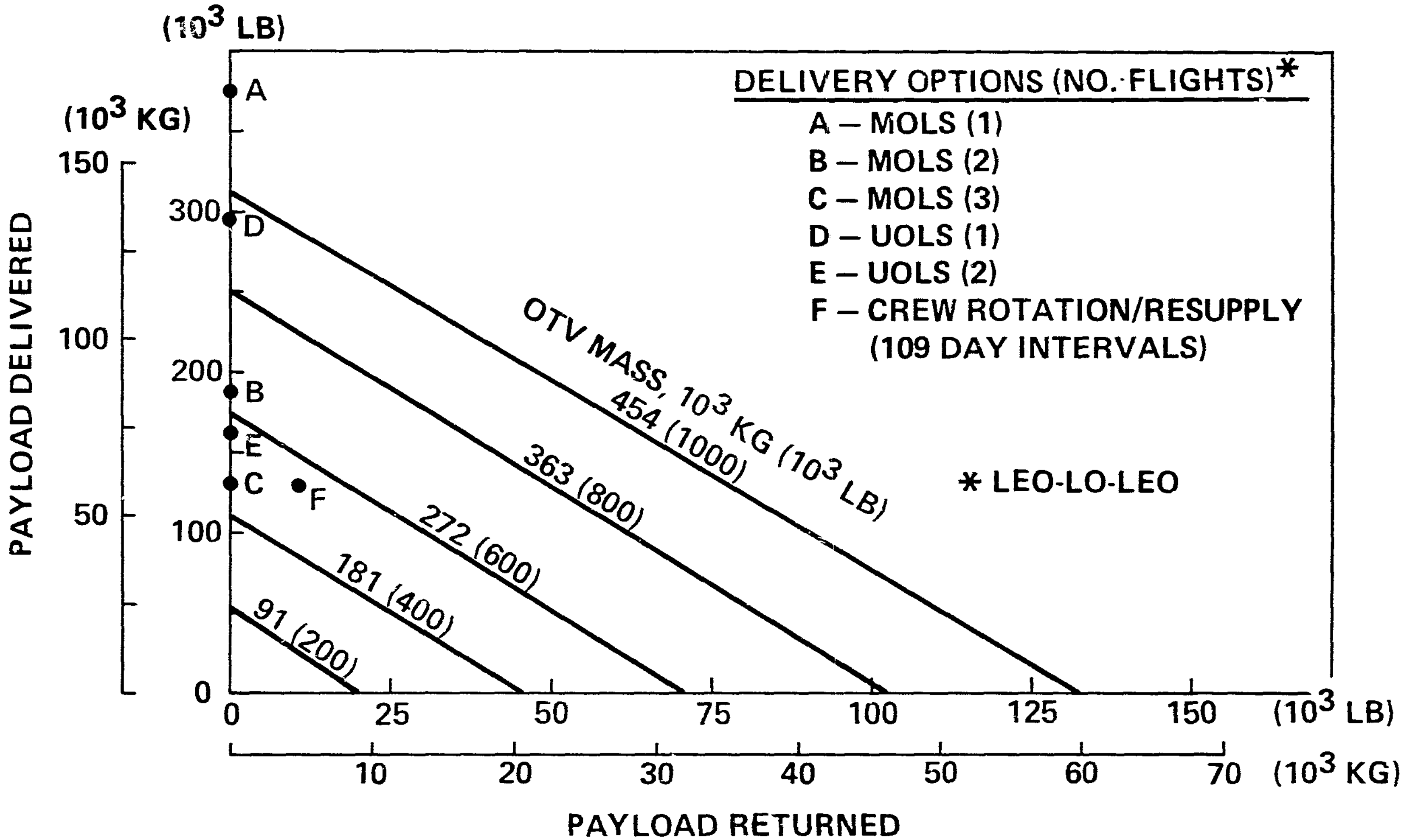


Figure 3.4-14. Single Stage LO_2/LH_2 OTV Capability for OLS

Table 3.4-11: Crew Involvement Summary Orbiting Lunar Station

PROGRAM / MISSION ELEMENTS / MISSION PHASE	STATION	ORBIT TRANSFER VEHICLE	PROPELLANT TANKER	CREW	CREW TRANSFER VEHICLE	LUNAR SORTIE VEHICLE	CARGO/ EXPER'T MODULES
EARTH TO EARTH ORBIT				PILOT SHUTTLE			
EARTH ORBIT	CHECKOUT ASSEMBLE FOR LUNAR TRANSFER	CHECKOUT ASSEMBLE TO PAYLOADS	DOCK TO OTV CONTROL/ MONITOR PROPELLANT TRANSFER		CHECKOUT	CHECKOUT	TRANSFER FROM SHUTTLE TO OTV
EARTH ORBIT TO LUNAR ORBIT		PILOT/ CONTROL IF ABOARD					
LUNAR ORBIT OPS	LIVE ABOARD	RENDEZVOUS WITH STATION		PERFORM ORBITAL MISSION	CHECKOUT	CHECKOUT	UTILIZE EXPERIMENTS FOR ORBITAL SCIENCE TRANSFER TO LSV OR OLS AS REQ'D
LUNAR DESCENT	STATION CREW STANDS BY TO ASSIST SURFACE CREW AS REQ'D					PILOT & PERFORM LANDING	
LUNAR SURFACE OPS	STATION CREW STANDS BY TO ASSIST SURFACE CREW AS REQ'D			PERFORM SURFACE MISSION		SURFACE CREW USES AS SURFACE SHELTER	DEPLOY & RELOAD AS REQUIRED
LUNAR ASCENT	STATION CREW STANDS BY TO ASSIST SURFACE CREW AS REQ'D					PILOT & PERFORM RENDEZVOUS	TRANSFER TO OLS AS REQUIRED
LUNAR ORBIT TO EARTH ORBIT		PILOT/ CONTROL IF ABOARD	CONTROL/ MONITOR ACQUISITION BY SHUTTLE		USE FOR EARTH RETURN AS REQUIRED	OPTIONAL USE FOR EARTH RETURN	TRANSFER TO OTV AS REQUIRED
EARTH ORBIT TO EARTH				PILOT SHUTTLE			TRANSFER TO SHUTTLE AS REQUIRED
DISPOSAL							

4. Cryogenic propellant transfer capabilities are required as follows:
   a. Tanker to OTV (unless OTV is not reusable and can be launched fully fueled into Earth orbit)
   b. Tanker to fluid module (FM)
   c. FM to LSV
   d. FM to surface mobility vehicle (SMV)
   e. FM to OLS cryogenic stores module
5. The LSV must be capable of docking to the OTV and to the OLS.
6. The LSV must be capable of docking with cargo modules docked to the OTV or OLS, undocking the cargo module from the OTV or OLS, and translating the cargo module to a suitable position for lunar landing.
7. In-orbit crew transfer is required, from a disabled or aborted LSV or OTV crew module to a rescue LSV or to the shuttle.
8. The LSV must be able to drop "down payload" cargo in order to accomplish a landing abort and return to lunar orbit.
9. The crew module of the OTV must be capable of separation from the OTV and subsequent propulsion delta V on the order of 400 m/sec (1,300 ft/sec) followed by a wait of up to several days for rescue.

Network Support—OLS mission will be largely autonomous and principal data return mode will be hard copy, tapes, and samples. Daily communications sessions will presumably be held while the OLS is line-of-sight with U.S. network sites. Orbit determination support will be required for station orbit trims and as a backup for rendezvous.

3.4.1.4.2.4 Earth Launch Requirements Summary

OLS delivery to lunar orbit requires 11 shuttle flights (delivery of OLS modules to Earth orbit) and 8 HLLV flights (delivery and refueling of OTV). Four shuttle flights and 7 HLLV flights annually are required to support the system.

3.4.1.4.3 Transportation Options Comparison and Evaluation

3.4.1.4.3.1 Size and Performance Comparison

Parametric performance maps were developed for all of the OTV candidates except the 1-1/2 stage

system which was on a point design basis. Plots for the alternate options with the superimposed delivery options associated with the station and crew rotation resupply are shown in figures 3.4-15 and 3.4-16.

OTV mass comparison for the candidate transportation systems is presented in figure 3.4-17. The OTV masses indicated are based on a payload that includes propulsion replenishment for the reference LSV (LO_2/LH_2 single stage). Staging gives the LO_2/LH_2 1-1/2 stage and common stage concepts a total mass advantage over the LO_2/LH_2 single stage. The LO_2/MMH common stage system is relatively massive due to its lower Isp. Sensitivity of OTV's to LTV options is indicated for the LO_2/LH_2 single stage OTV. The size comparison of figure 3.2-17 is applicable to this mission.

Performance maps were also developed for the LSV transportation candidates with the exception of the 1-1/2 stage concept which was analyzed on a point design basis. Performance plots for the alternate options are shown in figure 3.4-18 and 3.4-19. Payload for this delivery is the crew equipment module and the two modules containing exploration and science equipment.

Mass comparison of the LSV transportation concepts is presented in figure 3.4-20. Mass of the LO_2/MMH is approximately 11 300 kg (25,000 lbs) greater than the LO_2/LH_2 single stage. However, due to its greater density the LO_2/MMH system provides a total LSV length which is 3m (10 ft) shorter. The LSV options are shown in figure 3.4-21.

3.4.1.4.3.2 Earth Launch Requirements Comparison

Three OTV flights are required to deliver the OLS elements to lunar orbit. With 109 day resupply cycles, three OTV flights are also required on an annual basis.

The number of Earth launches required to deliver OTV hardware and fuel necessary to deliver the OLS element to initiate the mission is shown in figure 3.4-22. Several of the OTV candidates are dimensionally compatible with the space shuttle. All of the OTV concepts can be launched with the HLLV with a considerable reduction in number of Earth launches. Space shuttle flights are shown with two of the HLLV options since only a portion of the HLLV capability would be required to complete the delivery of the OTV systems.

In general, for those OTV candidates that can use either launch vehicle, the HLLV requires approximately one-third as many launches. Earth launches required to deliver OTV hardware and fuel necessary for the OLS annual requirements are shown in figure 3.4-23. Again, the HLLV requires only one-third as many launches as the space shuttle.

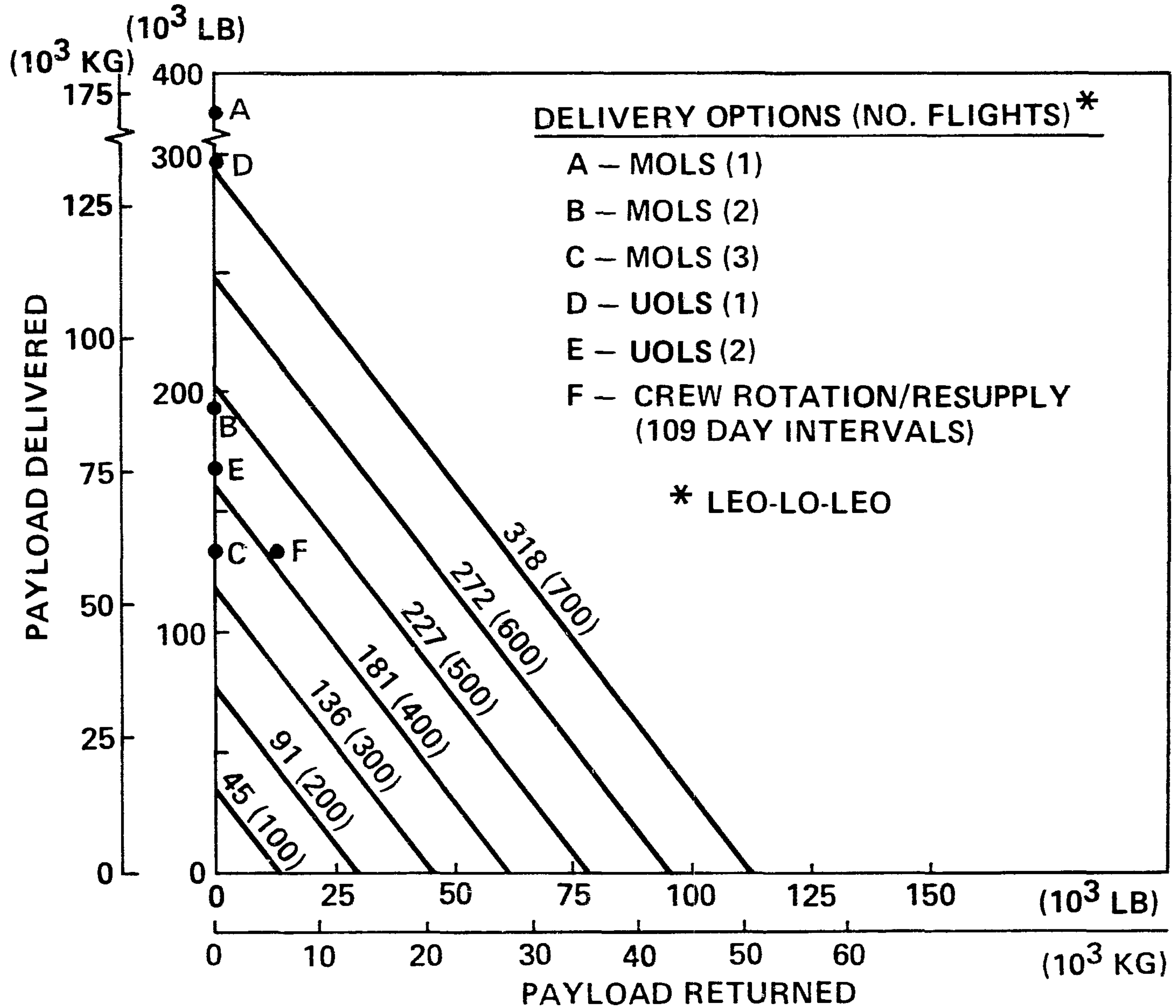


Figure 3.4-15. Common Stage LO_2/LH_2 OTV Capability for OLS

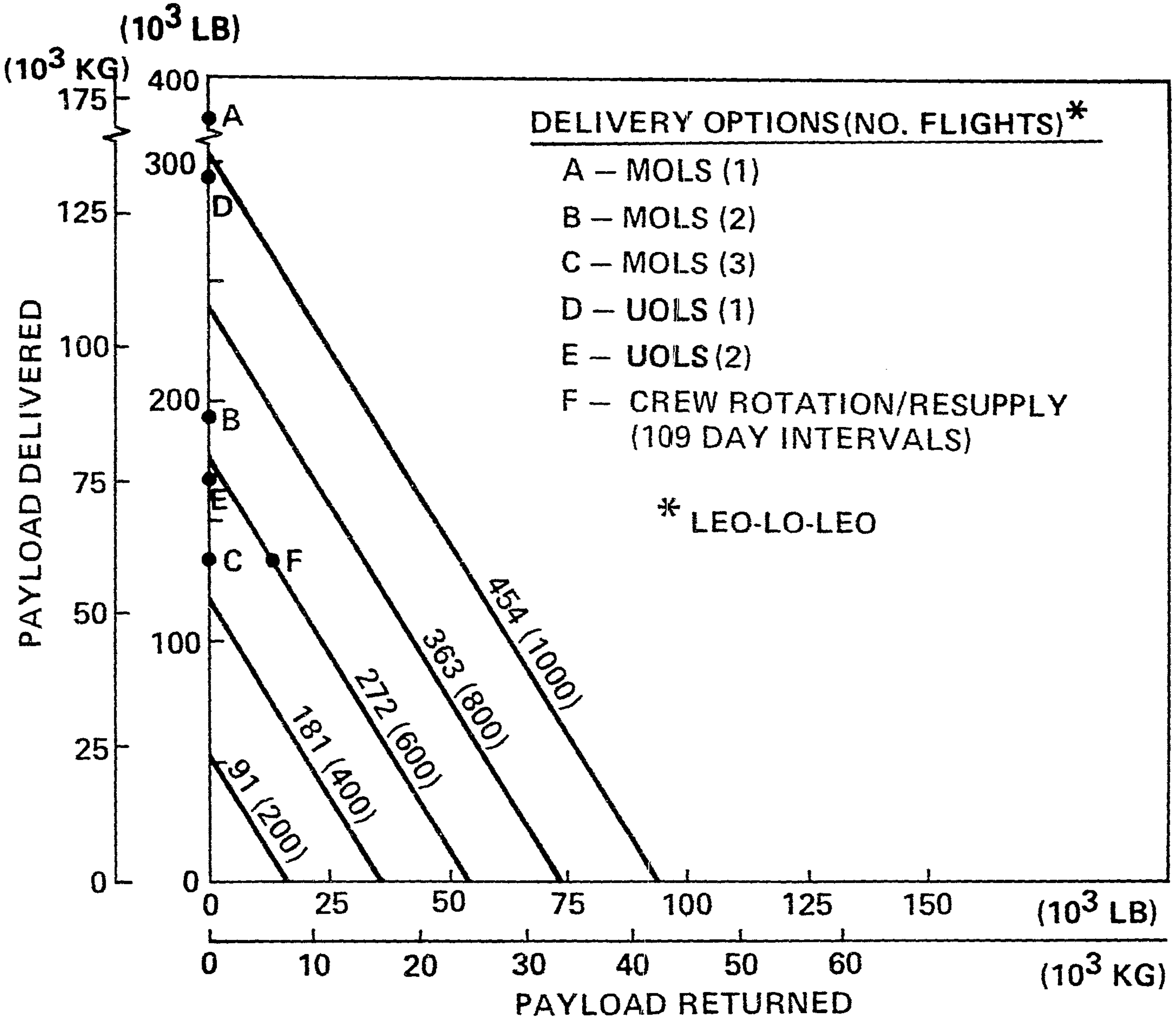


Figure 3.4-16. Common Stage LO_2/MMH OTV Capability for OLS

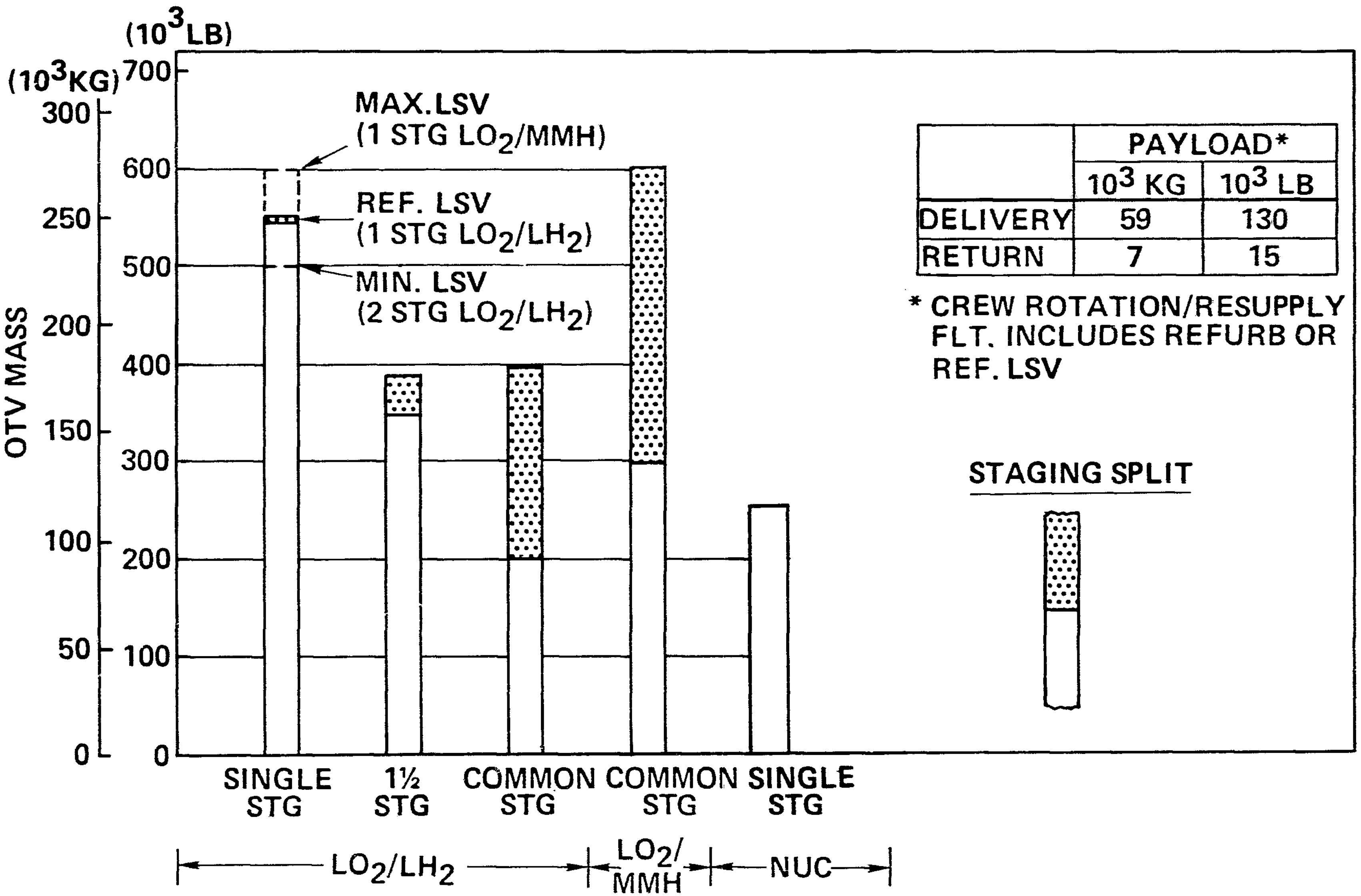


Figure 3.4-17. OTV Comparison for OLS

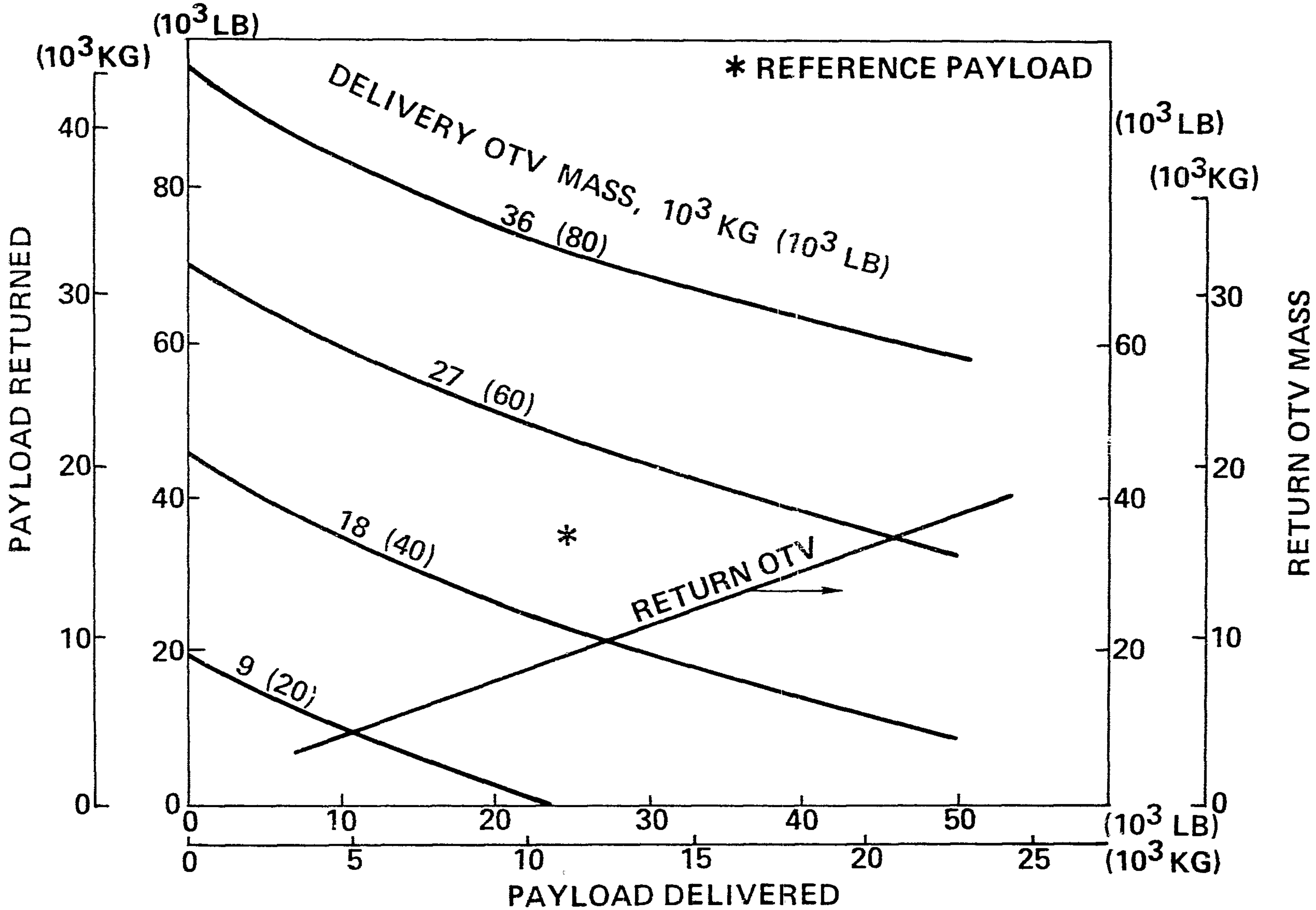


Figure 3.4-18. Two Stage LO_2/LH_2 LSV Capability for OLS

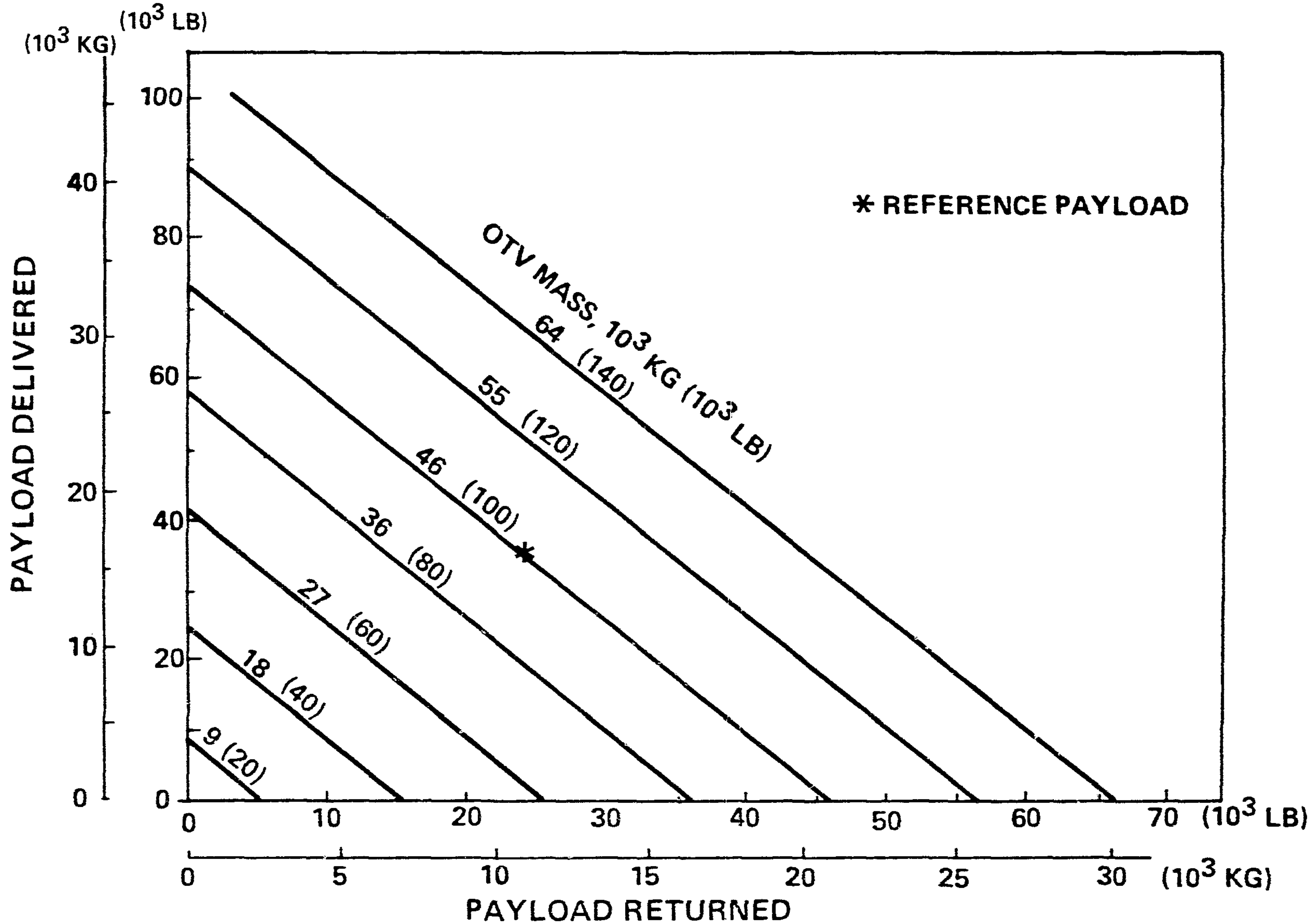


Figure 3.4-19. Single Stage LO_2/MMH LSV Capability for OLS

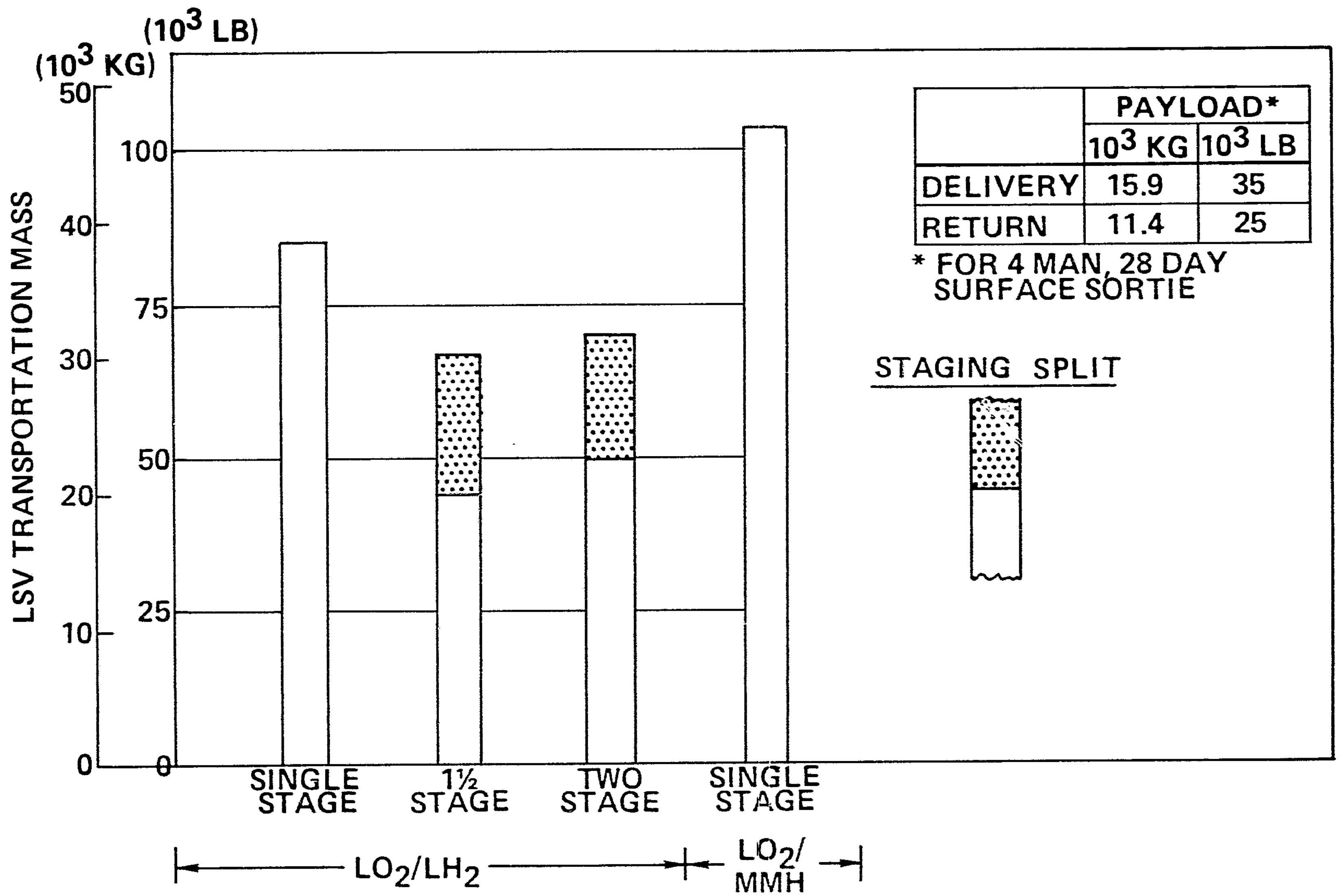


Figure 3.4-20. LSV Comparison for OLS

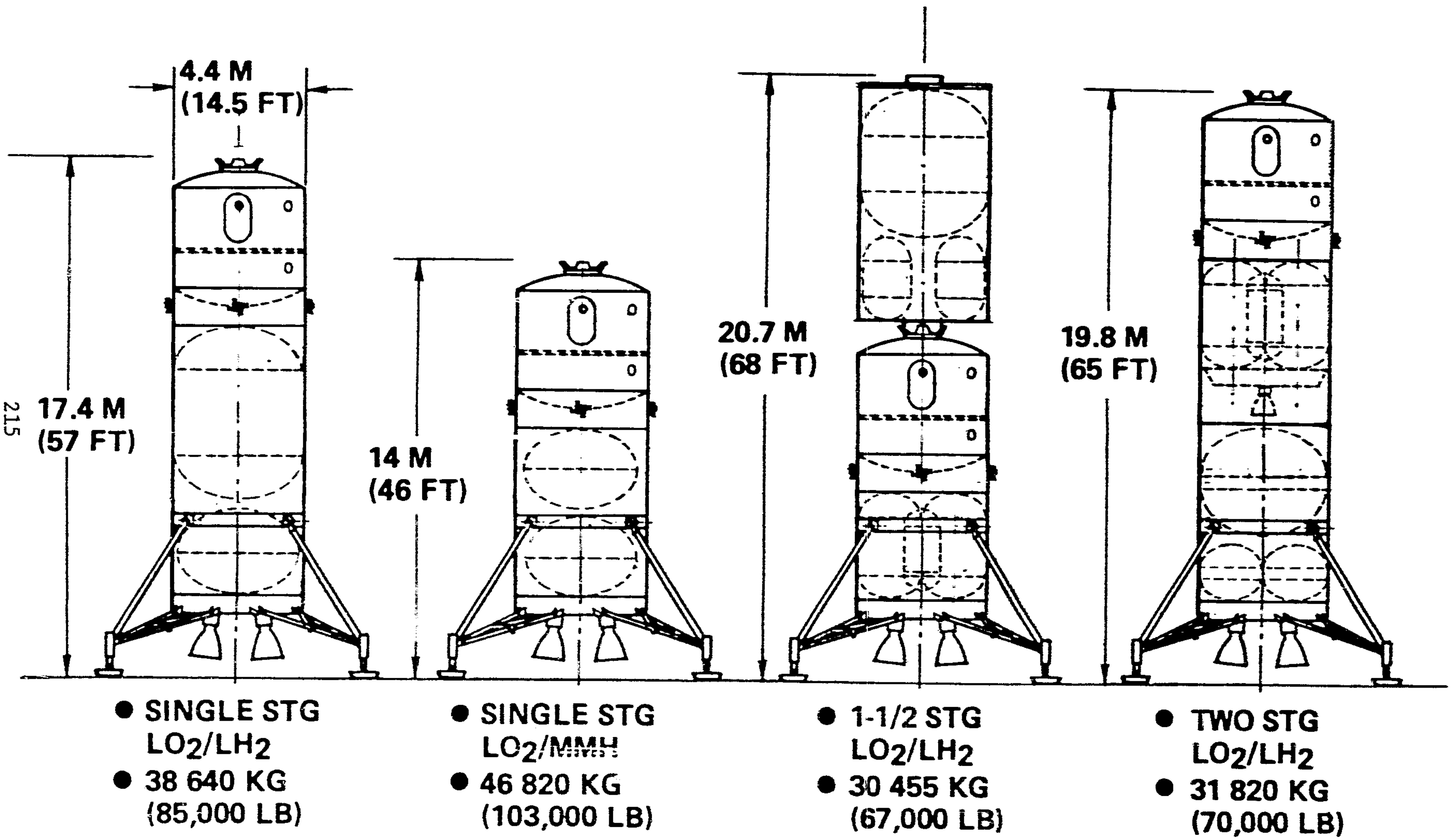


Figure 3.4-21 Representative LSV's for OLS Mission

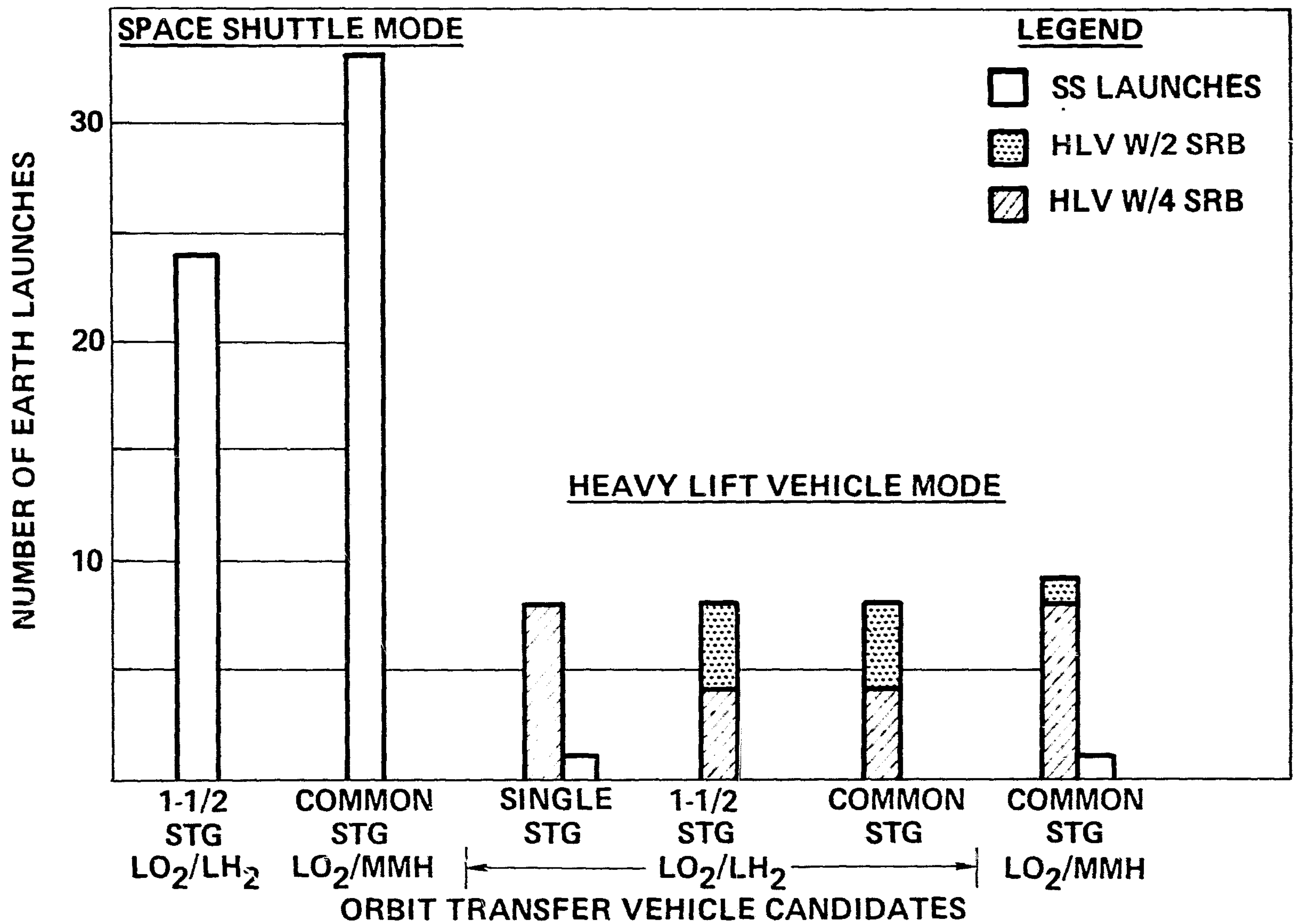


Figure 3.4-22. Earth Launches Required for OLS OTV System (Mission Start-Up)

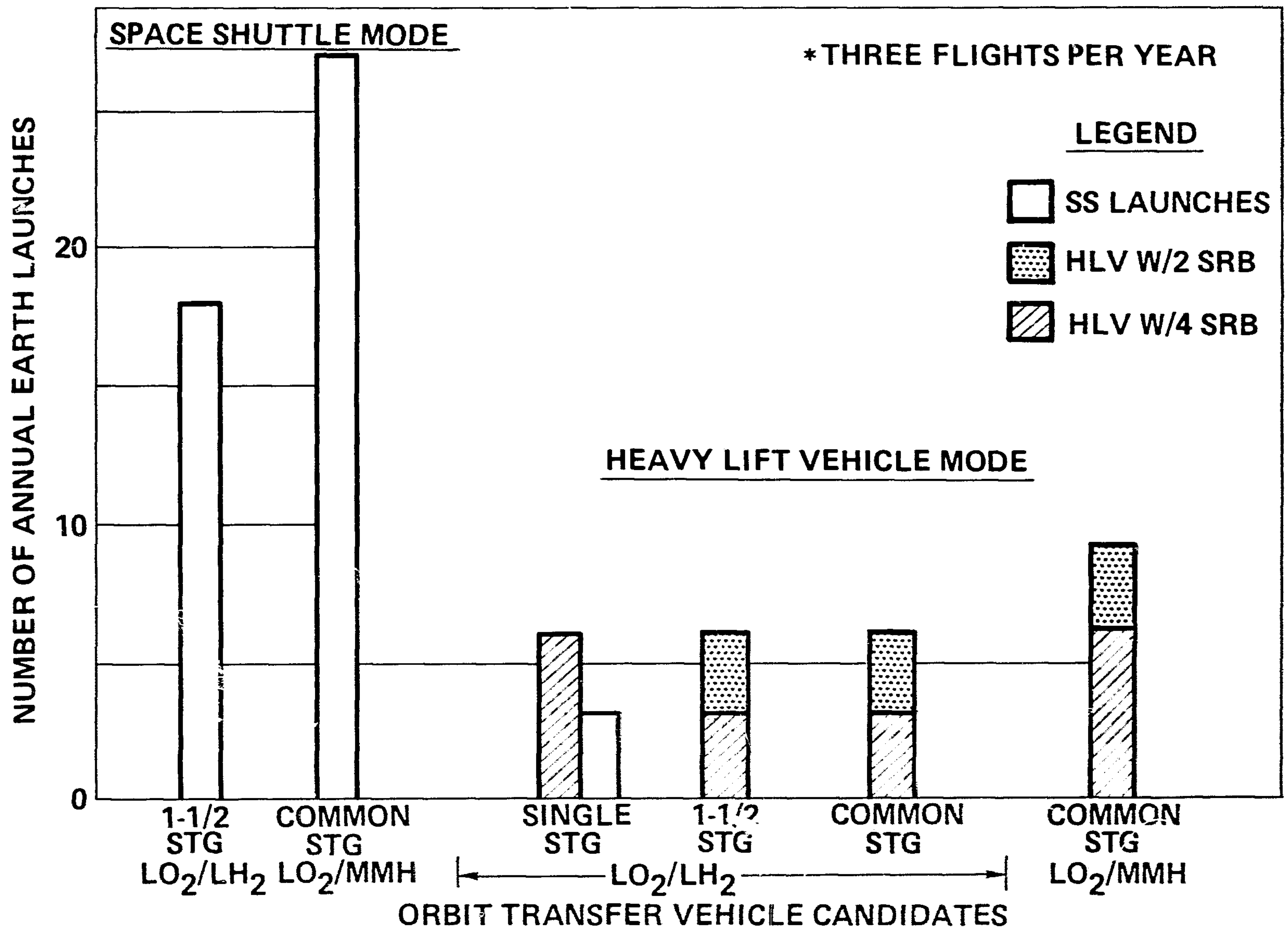


*Figure 3.4-23. Earth Launches Required for OLS OTV System (Annual Crew Rotation/Resupply) **

3.4.1.4.3.3 Operational Comparison

All modes investigated have essentially equivalent guidance, navigation, targeting, control, and network support requirements, with the exception that common-stage modes have an extra vehicle to track and recover. The docking comparison developed for the geosynchronous manned station mission (para. 3.2.1.4.3.3) applies. Nuclear modes were not investigated because of the operational problems found in the geosynchronous program analyses.

Lunar sortie vehicles present operational problems with (a) height of the crew from the lunar surface, (b) height of the vehicle c.g. at landing, (c) placement and handling of cargo. The single-stage vehicles appear the best from a c.g. height standpoint (about 6.4 meters, 21 ft) whereas the 1-1/2 stage vehicle has the best crew location. The 2-stage vehicle is very bad in both respects. The 1-1/2 stage vehicle presents a tank removal/disposal problem. All are comparable from the cargo handling standpoint.

3.4.1.4.3.4 Practicality Assessment

All modes and options investigated appear practical with the probable exception of the 2-stage LO_2/LH_2 LSV. Pro's and con's of the LSV's were discussed above, and are presented for OTV's in table 3.4-12.

Table 3.4-12. OTV Assessment

		ADVANTAGES					DISADVANTAGES	
		COMPATIBLE WITH SHUTTLE LAUNCH AND RECOVERY	NO STAGE-TO STAGE DOCKING	FULLY REUSABLE	ONLY ONE STAGE TO DEVELOP	NO LH_2	MULTIPLE DOCKING TO ASSEMBLE	TWO ACTIVE VEHICLES TO TRACK
LO_2/LH_2	SINGLE STAGE			x	x			
	1½ STAGE		x				x	
	COMMON STAGE			x	x			x
LO_2/MMH	COMMON STAGE		x	x	x	x		x

3.5 LUNAR SURFACE BASE PROGRAM

Two representative lunar surface base (LSB) missions, significantly greater in capability than the OLS surface sortie have been investigated: 1) a 6-man temporary base with durations up to six months; and, 2) 12-man semi-permanent or permanent base as illustrated in Figure 3.5-1 with mission durations up to five years.

The objectives of the 12-man LSB reference study were to conduct an evolutionary program leading to nearly permanent manned presence, while performing astronomical observations, deep drilling, remote explorations, local science and experiments on extraterrestrial resources utilization.

Accomplishment of this mission requires a variety of surface equipment, capability to rotate the crew, resupply capability, and transportation elements to move the equipment between the Earth and the moon.

The baseline LSB mission assumes no support from other major systems such as an orbiting lunar station or Earth orbit stations.

Principal reference for the mission is the Rockwell Lunar Base Synthesis Study of 1970 and 1971. Additional references include the Mimosa Study and a summer study of a lunar colony conducted at the NASA Johnson Space Center in 1972.

An additional point of interest on LSB missions involves recent studies investigating the feasibility of establishing space colonies. Some of these studies have assumed extensive use of lunar surface materials. With ideas of practical utilization of space receiving more emphasis, the 12-man LSB mission might place more emphasis on pilot plant operations processing indigenous materials rather than on astronomical observations. This study was conducted using the original 12-man LSB definition. It is believed that the postulated pilot plant payloads are comparable to the science payload definitions used. Insufficient definition of the former exists at present to enable a conclusive analysis of transportation requirements.

3.5.1 12-MAN SEMI-PERMANENT BASE

3.5.1.1 Mission Summary

3.5.1.1.1 General Description

The LSB surface elements primarily consist of a shelter for the crew and equipment, numerous mobility vehicles for surface transportation, and science equipment for astronomical and geological investigations. A variety of other modules and equipment serve in a support role.

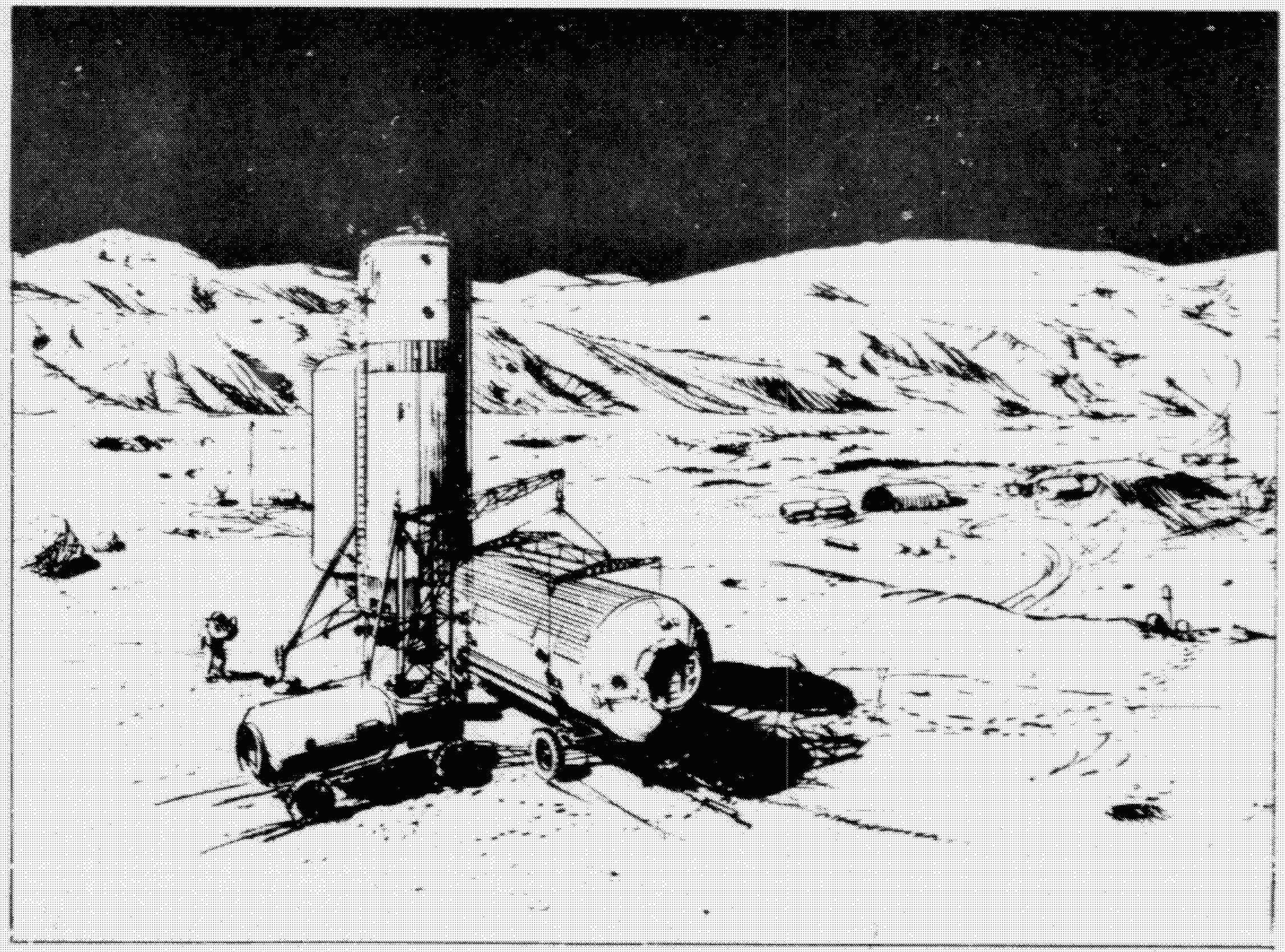


Figure 3.5-1. Lunar Surface Base

The LSB shelter consists of eight modules that provide quarters for the crew, command and communication center, maintenance facilities for base and mobility equipment, science labs and general base services such as medical, galley and recreation. All of these modules are 4.3m (14.5 ft) diameter with six being approximately 8.8m (29 ft) long and the others 11.3m (37 ft) long. The average mass is approximately 4 500 kg (10,000 lbs).

Four basic types of surface mobility vehicles as well as lunar flying vehicles are provided. The prime mover serves as the primary drive unit during the assembly of the LSB complex and on sortie missions. Each mobile power unit provides 3.5 kw from the Polonium 210 isotope/organic Rankine system and 1.2 kw emergency from a thermoelectric unit. Three of these units are dedicated to the shelter. The remainder are available for sortie missions or science use. The other self-powered units include the mobile crew shelter for long duration sorties away from the base and the utility trailer used for hauling bulk cargo, soil samples or the lunar flying vehicle (LFV). Mobility equipment not self-powered includes module transport trailers which are used to move modules and equipment that do not have wheels, a support operations equipment module that includes the hardware necessary for the lunar transport vehicle launch and landing operations and a mobile cargo supply module.

The major science payloads used in the LSB program include deep drilling provisions, radio telescopes, X-ray telescope and optical telescopes.

3.5.1.1.2 Mission Assumption and Constraints

Nominal mission assumptions and constraints are stated in table 3.5-1. The scientific and engineering experimental equipment indicated is representative and does not indicate a selection or recommendation for particular programs that might be carried out at a lunar surface base.

3.5.1.2 Mission Systems De iption

3.5.1.2.1 Mission Options

Not applicable.

3.5.1.2.2 Payload Descriptions

3.5.1.2.2.1 Base Shelter Modules

A summary description including quantity, size, and mass of the modules is presented in table 3.5-2. The total identified mass of the surface equipment, excluding transportation (cargo) modules, is approximately 118 000 kg (260,000 lb). Module diameters have been reduced from the referenced study value of 4.6m (15 ft) to 4.4m (14.5 ft) to comply with current shuttle ground rules.

Table 3.5-1. Lunar Surface Base Assumptions and Constraints

MISSION	OBJECTIVES	MISSION ASSUMPTIONS & CONSTRAINTS
LONG TERM MANNED LUNAR SURFACE OPERATION	EVOLUTIONARY PROGRAM LEADING TO PERMANENT, NEARLY SELF-SUFFICIENT MANNED PRESENCE, WHILE PERFORMING DEFINITIVE INVESTIGATION OF THE LUNAR SURFACE AND ASTRONOMICAL OBSERVATIONS • ASTRONOMICAL OBSERVATORY • DEEP DRILLING • REMOTE EXPLORATION • LOCAL SCIENCE • EXTRATERRESTRIAL RESOURCES UTILIZATION EXPERIMENTS	TWO BASE TYPES ARE HERE DEFINED FOR THE EVOLUTIONARY PROGRAM: TYPE — CREW — DURATION I — 6 — 6 MONTHS II — 12 — 5+ YEARS • CREW MODULE FOR MANNED ASCENT AND DESCENT WITH LIGHT EQUIPMENT, SURFACE SAMPLES, DATA • CREWMAN STAY TIME 6 TO 12 MONTHS • BACKSIDE CAPABILITY VIA RELAY SATELLITE • EXPLORATION EQUIPMENT AS FOLLOWS: TYPE — ROVER — OTHER I — 2-MAN — INTERMEDIATE DEPTH DRILL II — 4-MAN WITH AUXILIARY EQUIPMENT — DEEP DRILL + OBSERVATORY • OPTIONAL EXPERIMENTATION DIRECTED TOWARD UTILIZATION OF INDIGENOUS CONSUMABLES (LUNAR RESOURCES)

Table 3.5-2. *Lunar Base Equipment Definition*

ITEM	DESCRIPTION	QTY	UNIT WEIGHT*		UNIT SIZE (D x L)	
			KG	LBS	M	FT
SHELTER MODULES						
1. CREW & MEDICAL (CMM)	4 STATEROOMS,MEDICAL FACILITY	1	3,765	8,300	4.4 x 9.1	14½ x 30
2. CREW & OPERATIONS (COM)	4 STATEROOMS,COMMAND & COMMUNICATION CENTER	1	4,264	9,400	4.4 x 9.1	14½ x 30
3. SORTIE & TRANSIENT CREW (STCM)	4 STATEROOMS, BACK-UP GALLEY, BACK-UP COMM CENTER	1	3,992	8,800	4.4 x 9.1	14½ x 30
4. LAB & BACK-UP COMMAND (LBCM)	SCIENCE LABS, SCIENCE CONTROL CENTER	1	4,173	9,200	4.4 x 9.1	14½ x 30
5. GALLEY & RECREATION (GRM)	GALLEY/DINING,RECREATION, AIRLOCK/CLEAN ROOM	1	3,447	7,600	4.4 x 9.1	14½ x 30
6. BASE MAINTENANCE (BMM)	AIRLOCK/CLEAN ROOM MAINTENANCE & REPAIR	1	2,858	6,300	4.4 x 9.1	14½ x 30
7. DRIVE-IN GARAGE (DGM)	MOBILE EQUIPMENT STORAGE & MAINTENANCE	1	2,177	4,800	4.4 x 9.1	14½ x 37**
8. DRIVE-IN WAREHOUSE (DWM)	HOUSE MOBILE SUPPLY MODULE LSB SUPPLY STORAGE	1	2,314	5,100	4.4 x 11.3	14½ x 37**
	TOTALS		26,990	59,500		

*DRY

**DEVIATION FROM REFERENCE STUDY

The arrangement of the eight shelter modules is shown in figure 3.5-2. Three modules (CCM, COM, and STCM) are designed to nominally accommodate four men each and provide all the necessary crew services. In addition, each of these modules also provides a base service such as the medical facility, command and communications center, or backup galley and command center.

Other modules provide the science labs and control (LBCM), galley and recreation facility (GRM), maintenance facility for small base equipment (BMM), and drive-in warehouse (DWM) and drive-in garage (DGM) for mobile vehicle storage and maintenance.

All of the modules except the DWM and DGM are 4.4m (14.5 ft) diameter and 9.1m (30 ft) in length. These dimensions allow two modules to be launched to Earth orbit on a single shuttle flight. The DWM and DGM are both 4.4m (14.5 ft) diameter, but 11.1m (37 ft) in length in order to accommodate a prime mover and still be able to enter either of the adjacent modules. The center of gravity of each of the shelter modules is near its geometric centroid. Shelter modules have docking ports and are transported without containers.

The whole shelter is covered with lunar soil to provide thermal, radiation, and meteoroid protection.

3.5.1.2.2.2 **Operations Support Equipment**

Mobility Equipment—Four types of surface mobility vehicles and lunar flying vehicles are provided. The four surface mobility units are depicted in figure 3.5-3 performing a sortie mission away from the main base. Mobility equipment requires containerization for shipment.

The prime mover, illustrated in figure 3.5-4, is sized for 2-men/36-hour autonomous operation. The vehicle serves as the primary drive unit and control module on sortie missions and also accommodates soil and equipment handling devices.

Each mobile power unit provides 3.5 kw from the Polonium 210 isotope/organic rankine system and 1.2 kw emergency from a thermoelectric unit. Three of these units are dedicated to the shelter. The remainder are available for sortie missions or science use. The mobile crew shelter provides habitability on long-duration sorties away from the base and is sized for four men to satisfy the requirements of the deep-drill mission.

The utility trailer is used for hauling bulk cargo, soil samples, or the lunar flying vehicle (LFV). The LFV cart is similar in design to the utility trailer. The LFV provides the capability for two men to make short but fast sorties.

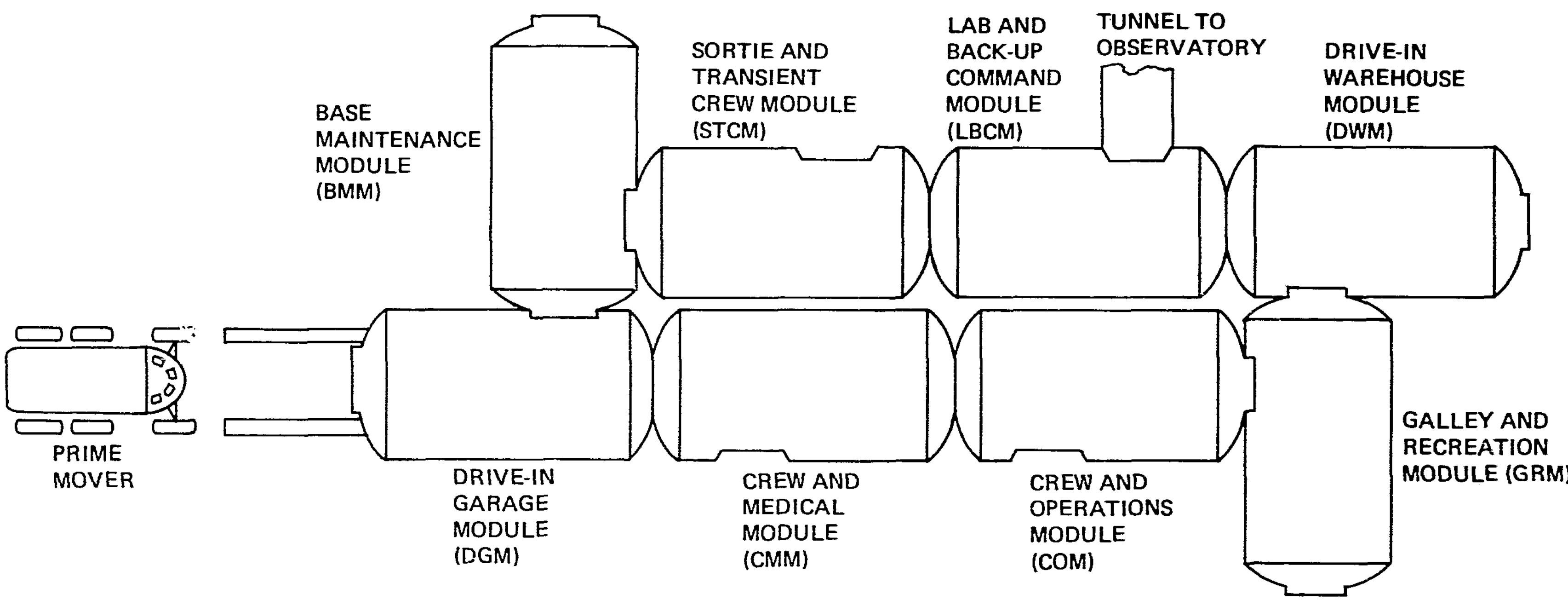


Figure 3.5-2: LSB 12-Man Shelter Complex

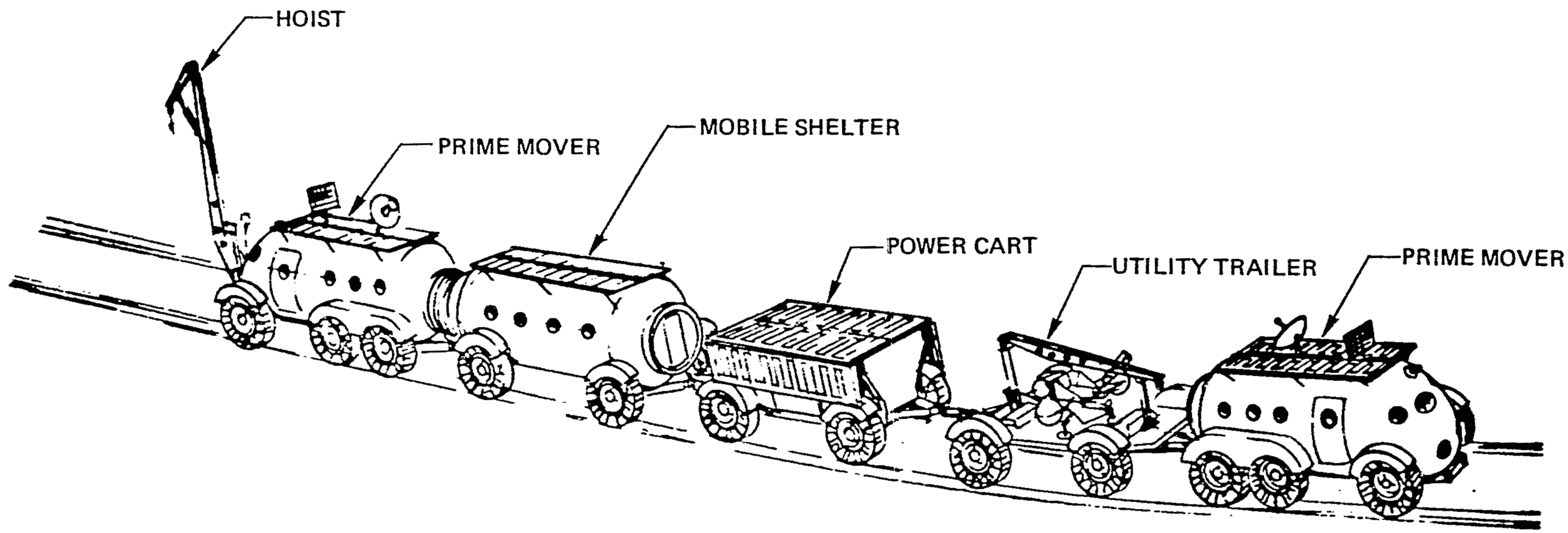


Figure 3.5-3 Sortie Mobility Elements

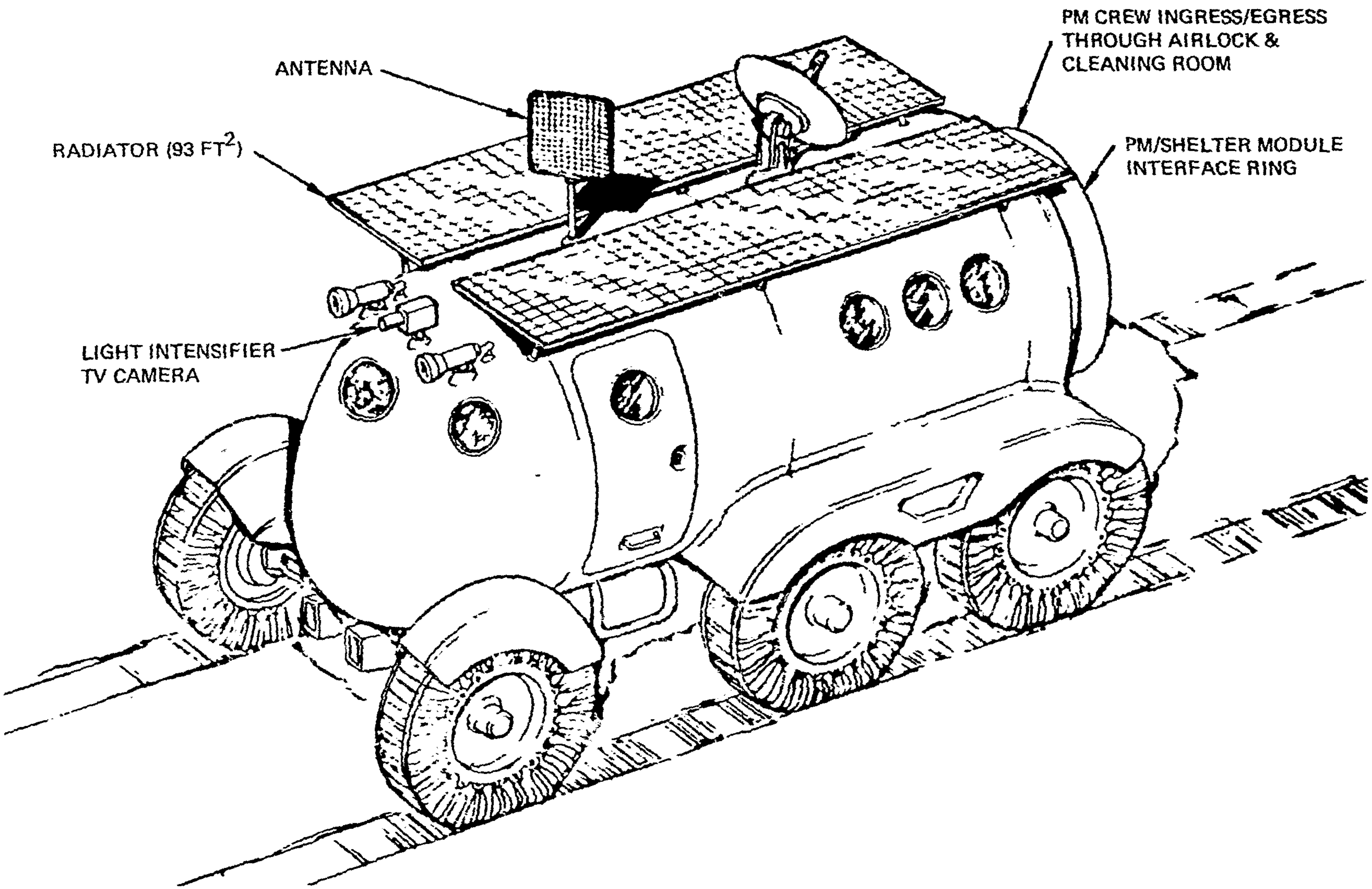


Figure 3.5-4: Prime Mover

Handling Equipment--Soil handling is accomplished by attaching a skiploader device to a prime mover. Tasks to be performed include excavation, covering, and leveling of terrain.

Auxiliary Modules Several modules serve mainly in a support role. The mobile cargo supply module can be taken to the landing site, loaded with supplies, and brought back to the base and housed in the drive-in warehouse.

The module transport trailer (MTT) is a flat-bed type unit used to move modules that do not have permanent wheels.

The support operations equipment module includes that hardware necessary for the lunar transport vehicle launch and landing operations. Included are navigation and beacon aids and liquification equipment to handle propellant boiloff from the lunar transport vehicle stationed on the surface for long periods.

Masses and sizes of this equipment are summarized in table 3.5-3.

3.5.1.2.2.3 Science Equipment

The major science equipment to be used for the LSB is shown in figure 3.5-5. Science equipment requires containerization for shipment.

Deep drilling for geological, geophysical, and geochemical investigations will include capability to obtain core samples from (100 to 1,000 ft) holes. Deep drilling is performed with a pressurized module that can be one of the cargo modules previously used to bring the LSB equipment to the Moon.

Four radio telescopes with frequency ranges of 0.3-1.0 MHz, 0.6-1.2 MHz, 1-15 MHz and 5-500 MHz are provided; they cover an area of 40 km x 65 km (25 mi x 40 mi).

The X-ray telescope has a 1m (40-in) aperture and a 10m (400 in) focal length. Operation of the unit is accomplished while wearing a pressure suit.

Both the 1.25m (50-in) and 2.5m (100-in) optical telescopes are enclosed within a protected dome. The 1.25m (50-in) telescope also is equipped with IR instrumentation. The 2.5m (100-in) diffraction-limited telescope consists of the following major assemblies: (1) dome, (2) horizontal tunnels, (3) 5m (200-in) flat pointing mirror, (4) a 2.5m (100-in) paraboloidal off-axis mirror, (5) smaller secondary mirrors, and (6) observation and instrumentation rooms. The latter two rooms can also be expended container modules used for transportation of equipment to the Moon.

Table 3.5-3. LSB Mobility and Support Equipment

ITEM	DESCRIPTION	QTY	UNIT MASS		UNIT SIZE (D x L)	
			KG	LB	M	FT
SELF MOBILITY EQUIPMENT						
1. PRIME MOVER (PM)	TRACTOR UNIT, 2-MAN ENCLOSED CAB	4	1 846	4,070	3.4 x 6.0	11 x 20
2 ELECTRICAL POWER MODULE (EPM)	3.5 KWe ISOTOPE/ORGANIC RANKINE	6	2 223	4,900	3.7 x 5.5	12 x 18
3. CREW SHELTER (MCS)	4 MAN CAPACITY	1	2 540	5,600	3.4 x 6.7	11 x 22
4. UTILITY TRAILER (UT)	CARGO MODULE OR FLAT BED	1	767	1,690	3.4 x 5.5	11 x 18
5. FLYER CART/FLYER (FC)	FLAT BED	1	862	1,900	3.4 x 5.5	11 x 18
	FLYERS	2	363	800		
HANDLING EQUIPMENT						
1. SOIL	SKIPLOADER	1	227	500	2.4 L	8 L
2. MODULES & EQUIPMENT	GENERAL PURPOSE HOIST	1	680	1,500	5.8 L	19 L
AUXILIARY MODULES						
1. MOBILE CARGO MODULE (MCM)	TRANSPORT CARGO FROM LANDING SITE TO BASE	1	454	1,000	2.7 x 5.5	9 x 18
2. MODULE TRANSPORT TRAILER (MTT)	TRANSPORT LARGE MODULES FROM LANDING SITE TO BASE	4	1 851	4,080	4.0 x 8.5	13 x 28
3. SUPP OPER EQUIP MOD (SOEM)	H_2/O_2 LIQUIFACTION, LTV LANDING AIDS	1	1 678	3,700	3.4 x 8.5	11 x 20
	TOTALS		36 060	79,470		

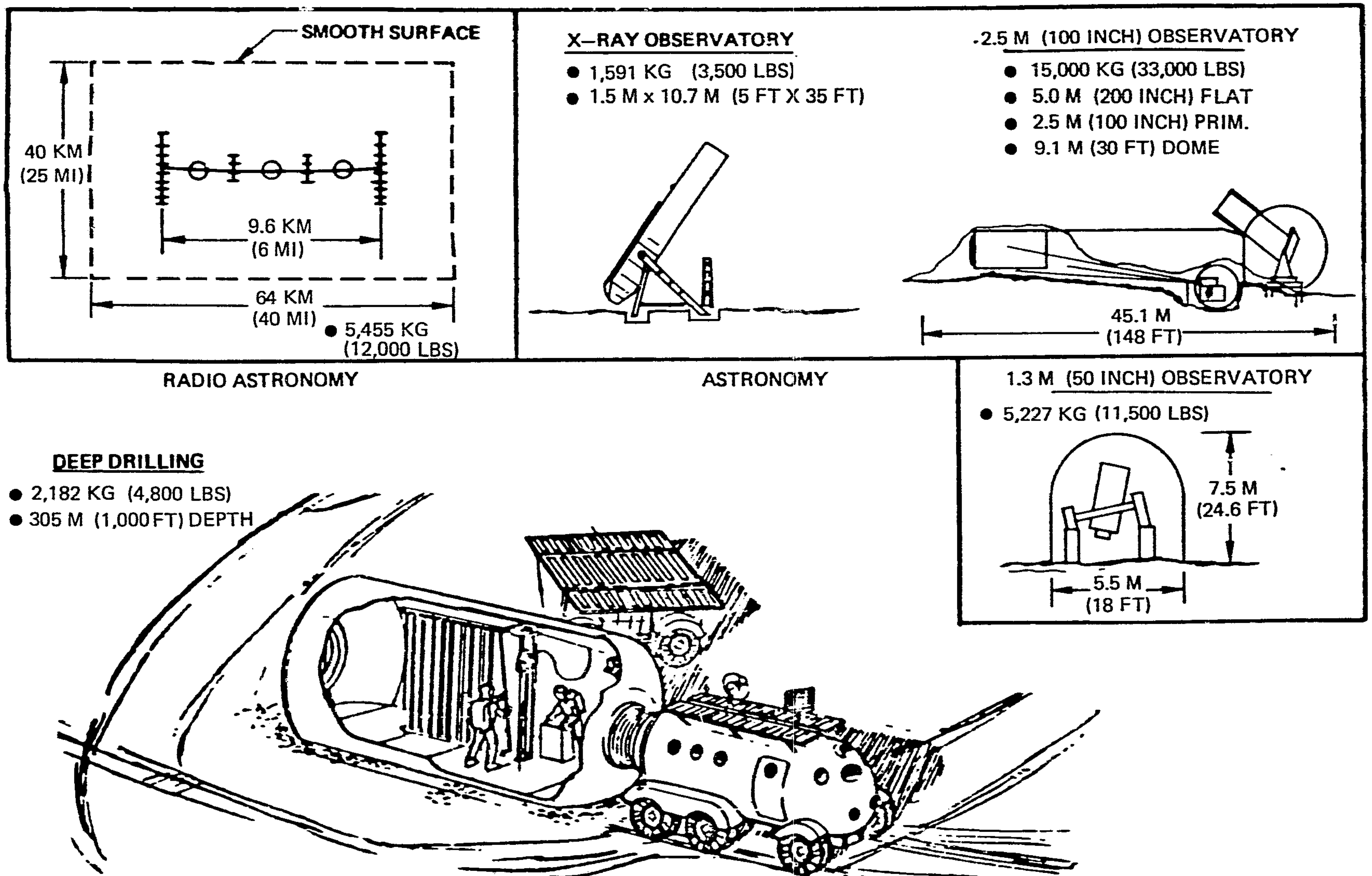


Figure 3.5-5. LSB Science Equipment

Masses and sizes of science equipment are summarized in table 3.5-4.

3.5.1.2.2.4 Initial Supplies

Supplies for the first 5 months (27 manmonths) of buildup have a mass of 15 500 kg (34,300 lb) as summarized in table 3.5-5.

3.5.1.2.2.5 Crew Transfer and Resupply Payloads

Resupply intervals of 164 days were defined by the reference study. Crew rotation cycles were allowed to extend to 328 days since the shelter is covered with soil to reduce the radiation, crewmen are operating under a gravity field and are not confined to small quarters. Consequently, every 164 days a combination crew rotation/resupply flight will be flown with one-half of the crew rotated.

Crew Transport Vehicle–The primary function of the CTV is to provide quarters for the crew during transits between Earth and lunar orbit. The CTV is normally transported between the orbits by an orbit transfer vehicle (OTV). The CTV consists of a module providing shirt sleeve environment for the crew and a portion of the required supporting equipment. An unpressurized equipment module is included for the remainder of the support equipment.

This module is sized to accommodate a crew of six, the normal crew rotation complement. This vehicle need not be sized for the entire LSB crew of 12 in the event of emergency evacuation of the base, because the CEM/LTV that brings the crew up from the base to lunar orbit can be refueled and transport at least 6 crewmen back to Earth orbit. The CTV crew module must also accommodate the lunar payload to be returned to Earth (i.e., film, tapes, surface samples, etc.).

The equipment module houses electrical-power sources, cryogenic tankage, and a small propulsion system. The purpose of the propulsion system is to provide the capability for the CTV to achieve a 1-day Earth parking orbit in the event the OTV fails to operate during Earth-orbit insertion.

Crew/Equipment Module (CEM)–The CEM is part of the lunar transport vehicle and provides quarters for the crew and the required support equipment during transits between lunar orbit and the LSB. The CEM of the LTV is similar to the CTV used between Earth and lunar orbit but with several differences. One difference is that the CEM only needs provisions for 1 to 2 days; the CTV must have supplies for an Earth return of up to 10 days. The second difference is that the CEM must accommodate the entire LSB crew of 12 in event of a base evacuation; the CTV only needs to house the normal crew-rotation complement of 6. In addition to the crew, the CEM must also

Table 3.5-4. Lunar Surface Base Science Equipment

ITEM	DESCRIPTION	QTY	UNIT MASS		UNIT SIZE (D x L)	
			KG	LBS	M	FT
SCIENCE						
1. DRILL COVER MODULE	SHIRT SLEEVE ENVIRON. USE EMPTY CARGO MODULE	1				
2. DRILL EQUIPMENT	SMALL, MEDIUM, 1000 FT DRILLS & MISC SUPP EQUIP	1 SET	2 177	4,800	3.0 L	10 FT L
3. REMOTE SORTIE EQUIP	SCIENCE	1 SET	1 542	3,400	N/D	
4. 50 IN. AST. OBSERVATORY	TELESCOPE, DOME, INSTR	1	5 239	11,550	4.9 DOME 1.5 x 3.0	16'D DOME 5 x 10 SCOPE
5. X-RAY AST. OBSERVATION	TELESCOPE, INSTR.	1	1 588	3,500	1.5 x 10.7	5 x 35 SCOPE
6. RADIO AST. OBSERVATORY	ANTENNA, INSTR	1	5 443	12,000	N/D	N/D
7. 100 IN. OBSERVATORY	TELESCOPES, DOME, INSTR.	1	14 969	33,000	9.1 DOME 5.1 FLAT 2.5 PRIM	30'D DOME 200 IN. FLAT 100 IN. PRIM
8. 100 IN. OBSERV. SHELTERS	2 REQUIRED – USE EMPTY CARGO MODULE					

Table 3.5-5. Initial Supplies

ITEM	DESCRIPTION	MASS	
		KG	LBS
FLYER PROPELLANT	LO_2/LH_2	1 361	3,000
EPS SPARES		386	850
PRIME MOVER SPARES		2 132	4,700
CREW CONSUMABLES	FOOD, CLOTHES, ETC.	10 074	22,210
EVA SPARES		853	1,880
A&CS SPARES		227	500
COMM & DATA SPARES		381	840
MEDICAL		145	320
	TOTALS	15 559	34,300

transport the lunar payload that is to be returned to Earth. It should also be noted that since the CEM includes the G&N and information subsystems for the LTV, the CEM must be included with the LTV when it is flown unmanned.

Resupply Module (RM)—The RM will contain both LSB crew consumables and surface expendables. On a normal resupply mission, only supplies will be landed at the LSB. Therefore, two RM's are provided in order to provide a reasonable vehicle c.g. Assuming the use of a 4.4-meter (14.5 ft) diameter module and an average packing density of 160 kg/m^3 (10 lb/ft^3), module lengths of 5 meters (16.5 ft) are required.

LTV Resupply—Replenishment of the lunar transport vehicle (LTV) used to transfer payloads between lunar orbit and the lunar surface is also required. The representative option employs a fully reusable single stage LO_2/LH_2 LTV; resupply consists of propellants and auxiliary fluids.

Table 3.5-6 summarizes crew rotation and resupply requirements.

Table 3.5-6. LSB Crew Rotation and Resupply Payloads

Payloads	Description	Mass		Size (dia x len)	
		10^3 kg	10^3 lb	meters	feet
Lunar Ascent					
• Crew and equipment modification (CEM)	Sized for 6 men, 6 days. Emergency 12 men returned to Earth	4.0	8.8	4.4 x 4.6	14.5 x 15.0
• Crew, equipment, and reserves	Normal–6 men	1.0	2.2		
	Emergency–12 men	(1.4)	(3.1)		
• Consumables	For crew supply	0.2	0.4		
• Science	Samples, film, tape	0.4	0.9		
Total		5.6	12.3		
Lunar Descent					
• Ascent P/L plus	See above	2.7	6.0	4.4 x 4.6	14.5 x 15.0
• Resupply module	Dry–2 required	16.8	37.0		
• Base consumables	169 days	1.3	2.9		
• CEM consumables					
Total (ascent + decent)		26.4	58.2		
Lunar Orbit–Earth Orbit					
• Crew transfer vehicle	Sized for 6 men	5.3	11.7	4.4 x 4.6	14.5 x 15.0
• Crew, reserves and consumables		2.1	4.6		
Total returned		7.4	16.3		
Earth Orbit– Lunar Orbit					
• Return payloads plus	Dry				
• Propellant module		3.8	8.4	4.4 x 10.7	14.5 x 35.0
• Propellant		37.2	82.0		
Total delivered		70.8	156.0		

3.5.1.2.2.6 Consumables

Consumables requirements were included in table 3.5-6. Details of base consumables requirements are given in table 3.5-7. The following consumables use assumptions were made:

O_2:	From water electrolysis
N_2:	From N_2H_4 dissociation
Water:	Two closed loops are used one for wash water and another for the remainder of the water. Water is also produced in the CO_2 reduction process.
H_2:	For CO_2 reduction; from N_2H_4 dissociation
Clothing:	Expendable
Waste:	Toilet, dry john Trash, vacuum dry—bury other waste
Food:	Wet and dry

Table 3.5-7. LSB Supplies for 12 Men — 180 Days

	KG	LB
ATMOSPHERE AND CREW SYSTEMS		
CONSUMABLES/EXPENDABLES	9,700	21,036
SPARES	228	500
MEDICAL	137	300
EVA SPARES		
SUITS, APLSS, ETC.	817	1,800
SCIENCE SUPPORT		
SORTIE	216	477
ASTRONOMY	445	980
COMMUNICATION AND DATA		
SYSTEMS	380	840
VHF RELAY LINKS	455	1,000
ELECTRICAL POWER		
ISOTOPES AND HOLDERS (6 UNITS)	2,180	4,800
SYSTEMS (6 UNITS)	530	1,170
MOBILITY SYSTEMS		
SPARES (4 PRIME MOVERS)	1,810	4,000
	16,774	36,903

3.5.1.2.2.7 Mass Summary

Mass delivery requirements as identified in source data are summarized in table 3.5-8. These values do not include growth allowances.

Table 3.5-8. Mass Summary

ITEM	MASS	
	10^3 KG	10^3 LB
INITIAL DELIVERY		
• SHELTER MODULES	27.0	59.5
• MOBILITY EQUIPT	36.1	79.5
• SCIENCE	31.0	68.3
• INITIAL SUPPLIES	15.6	34.3
TOTAL	109.7	241.6
TYPICAL RESUPPLY		
• DELIVERED TO LUNAR SURFACE	26.4	58.2
• ASCENT	5.6	12.3

3.5.1.2.2.8 Pickup Points and Transportation Constraints

Shelter modules are suitably configured for transportation and handling; they will be handled like modular space station elements (see para 3.2.1.2.2.6). Base equipment requires packaging in equipment delivery modules (EDM's).

The equipment delivery module (EDM) quantity was established based on estimated equipment sizes stated in the reference study. Using this approach, a total of 28 EDM's were required to deliver the surface base equipment.

This number is required because there are many units of equipment so large that only one unit can be accommodated by a single EDM.

Since in most cases the major units do not occupy the entire length or diameter of the EDM, the base supplies and other science equipment can be accommodated by these same EDM's.

The type of equipment included in each EDM is presented in table 3.5-9. The module number listed is a preliminary estimate of the delivery sequence. The primary considerations in establishing this order was that shelter, electrical power, and mobility systems must be available early to allow efficient construction of the base. It should also be noted that equipment other than major units can be switched between modules if necessary.

Table 3.5-9. Lunar Surface Base Delivery Modules

MODULE NO.	EQUIPMENT	WEIGHT (K) KG	WEIGHT (K) LB	VOLUME UTILIZATION FACTOR
1	CREW & COMMAND MOD CONSUMABLES SPARES	5.9	13.0	1.0
2	DRIVE-IN GARAGE MOD PRIME MOVER HANDLING EQUIP	5.4	12.0	1.0
3	DRIVE-IN WAREHOUSE MOD POWER UNIT FUEL J BOX	5.5	12.2	1.0
4	CARGO MODULE MTT MCM CONSUM	5.4	12.0	1.0
5	CARGO MODULE SUPP OPER EQUIP CONSUM 100" OBS (INSTR)	6.4	14.0	0.8
6	GALLEY & REC MOD CONSUMABLES	4.8	10.6	1.0
7	CARGO MOD PWR UNIT/FUEL	4.0	8.9	0.7
8	CREW & MED MOD CONSUMABLES SPARES	5.3	11.6	1.0
9	CARGO MODULE PWR UNIT/FUEL EPS SPARES	4.1	9.4	0.7
10	CARGO MODULE DRILLS MTT	5.7	12.6	0.8
11	CARGO MODULE P MOVER CONSUM	4.0	8.9	0.7
12	LAB & BU COMM MOD	4.1	9.0	1.0
13	CARGO MODULE PWR UNIT/FUEL CONSUM	4.9	10.9	0.7
14	BASE MAINT MOD CARGO MODULE PWR UNIT/FUEL CONSUM	2.7	6.0	1.0
15	CARGO MODULE POWER UNIT/FUEL CONSUM	4.9	10.9	0.7

NOTE: ALL MODULES ARE 4.4 x 9.1M (14½ x 30 FT)

Table 3.5-9. Lunar Surface Base Delivery Modules (Cont)

MODULE NO.	EQUIPMENT	WEIGHT (K) KG	WEIGHT (K) LB	VOLUME UTILIZATION FACTOR
16	CARGO MODULE P MOVER & SPARES REMOTE SORTIE EQUIP CONSUMABLES	5.7	12.5	0.9
17	CARGO MODULE 100" OBS (DOME & MISC STRUCT)	5.9	13.0	0.6
18	CARGO MODULE 100" OBS (2 PART-200" FLAT MIRROR)	6.4	14.0	0.7
19	CARGO MODULE 100" MIRROR, SEC MIRRORS, MOUNT/ CONTROLS)	6.4	14.0	1.0
20	CARGO MODULE PWR UNIT/FUEL 50" OBS (DOME + INSTR)	5.4	11.9	1.0
21	CARGO MODULE P MOVER & SPARES CONSUM X RAY OBS (ASPELTT)	5.2	11.4	1.0
22	SORT & TRANS CREW MOD SPARES	3.9	8.6	1.0
23	CARGO MODULE UTIL TRAIL/FLYER/PROP 50" OBS (INSTR & CONT) RADIO T (1-15 MHz)	5.9	13.1	0.9
24	CARGO MODULE MCS EVA SPARES/CONSUM CONSUM	5.5	12.1	0.9
25	CARGO MODULE 50" OBS (SCOPE & YOKE) MTT	6.4	14.0	1.0
26	CARGO MODULE RADIO T (.3-1.0 MHz)	5.0	11.0	0.8
27	CARGO MODULE FLYER CART/FLYER/PROP REMOTE SORTIE EQUIP RADIO T (0.6-1.2 & 5-500 MHz)	4.8	10.6	0.8
28	CARGO MODULE MTT X-RAY T (PRIMARY & SUPP EQUIP)	5.0	11.0	1.0

NOTE: ALL MODULES ARE 4.4 x 9.1m (14½ x 30 FT)

The masses specified for each module include that of the primary structure of the EDM plus the payload. The average mass of the loaded EDM's is 4 500 to 5 500 kg (10,000 to 12,000 lb). The minimum is 3 900 kg (8,600 lb) and the maximum is 6 400 kg (14,100 lb). These values do not include growth allowance.

The volume factor (VF) indicated is an estimate of the ratio of equipment volume to total volume of a given EDM. Sixty percent of the EDM's have a volume factor of 0.9 or greater.

EDM's with 0.6 to 0.7 VF generally contain large major units of equipment or dome structures for the observatories. (Large sections minimize construction effort.) These low volume-factor EDM's, however, could be used to carry some of the equipment of the EDM's weighing 5 900 to 6 300 kg (13,000 to 13,900 lb) if that equipment could be designed in small units. This approach would have the benefit of reducing the maximum EDM weight to approximately 5 500 kg (12,000 lb) at the expense of increased construction time.

3.5.1.2.3 Transfer and Storage

The concept for handling payload modules by the LTV while in order and on the lunar surface is shown in figure 3.5-6. The main feature used in the payload handling is a module docking adapter (MDA) mechanism located at the base end of the LTV. Either two or four docking adapters can be incorporated.

Unloading of payloads on the lunar surface consists of having the MDA lower the payload to a position where it can be placed on a module transport trailer (MTT). Final removal of the payload module from the LTV requires the use of the prime mover with the hoist.

Propellant transfer is required from the propellant module to the LTV in lunar orbit. The LTV's are stored on the lunar surface at the base between resupply missions. The reference study assumed that electrically-powered propellant recovery and reliquification equipment would be used on the lunar surface to recover and recycle LTV boiloff.

3.5.1.2.4 Orbital Assembly, Maintenance, and Modification

Not applicable.

3.5.1.3 Transportation Requirements

3.5.1.3.1 Payload Delivery Points

Payload delivery points are summarized in table 3.5-10.

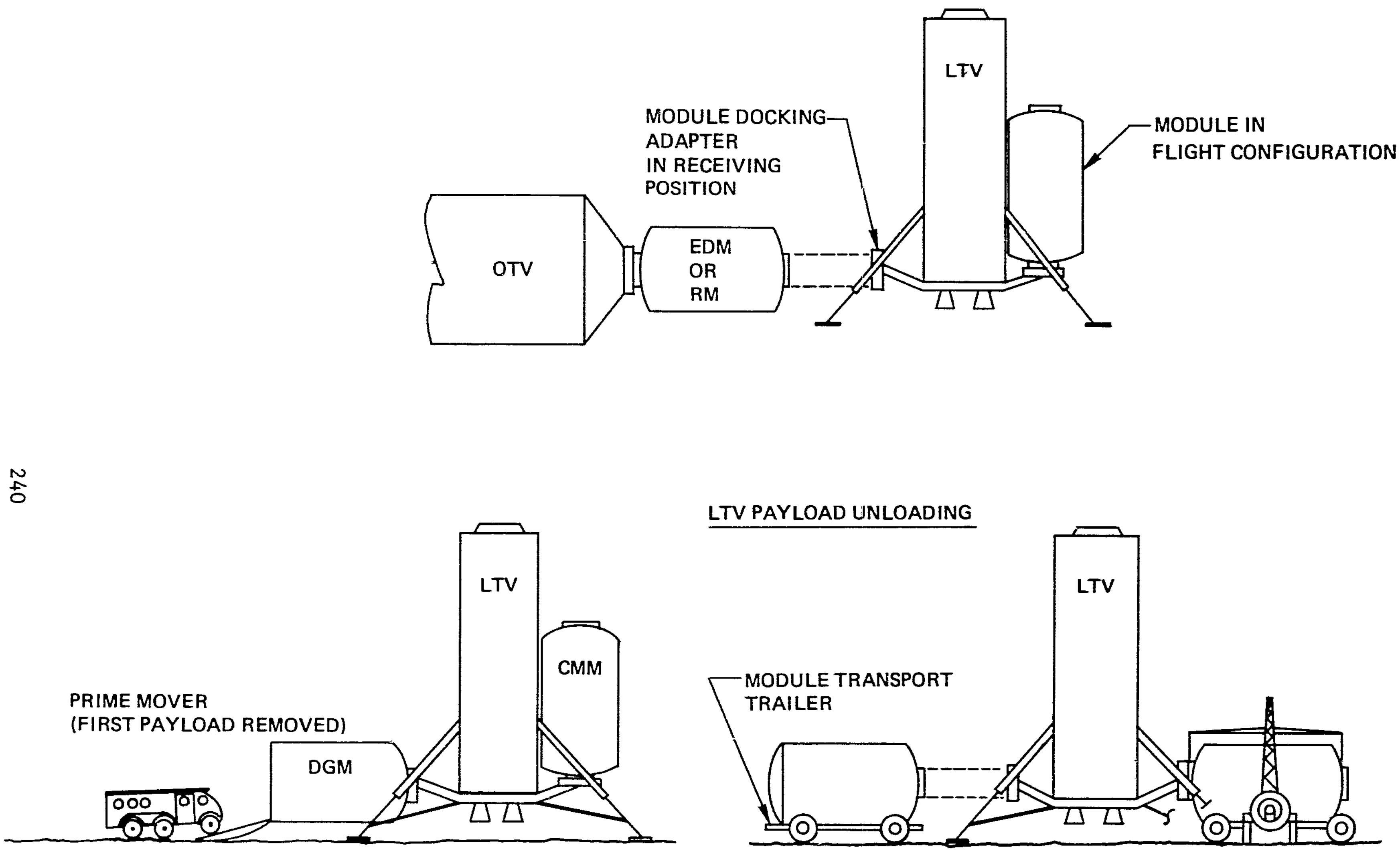


Figure 3.5-6 Payload Handling Concepts

Table 3.5-10. *Lunar Surface Base Delivery Points*

BASE ELEMENTS / DELIVERY POINTS	OLS (IF PRESENT)	SURFACE BASE MODULES	ORBIT TRANSFER VEHICLE	OTV TANKER & PROPELLANT	CREW	CARGO & EXPERIMENTS	CREW TRANSFER VEHICLE (CTV)	LUNAR TRANSPORT VEHICLE (LTV)
TO EARTH ORBIT	SS OR HL	SS OR HL	SS OR HL	SS OR HL	SS	SS	SS OR HL	SS OR HL
EARTH ORBIT TO LUNAR ORBIT	OTV	OTV	SELF		OTV	OTV	OTV	OTV
LUNAR ORBIT TO LUNAR SURFACE		LTV			LTV	LTV	SELF	SELF
LUNAR SURFACE TO LUNAR ORBIT					LTV	LTV	SELF	SELF
LUNAR ORBIT TO TO EARTH ORBIT			SELF		OTV OR LTV OR CTV	OTV	SELF	SELF
EARTH LANDING		LEFT ON MOON		SS	SS OR CTV	SS OR CTV	SELF	SS
DISPOSAL	LUNAR CRASH		REUSED; CONTROLLED EARTH ENTRY AT END OF LIFE	REUSED		REUSED OR LEFT ON MOON	REUSED	REUSED

NOTE: SS = SPACE SHUTTLE HL = HEAVY LIFT

3.5.1.3.2 Payload Delivery Options

3.5.1.3.2.1 Configurations

For base delivery, EDM's and shelter modules are delivered one or two at a time, depending on length, by the shuttle. They are arranged in combinations in Earth orbit as appropriate to the capabilities of the OTV/CTV transportation system.

Crew rotation/resupply payloads are grouped into a single assembly for a joint crew rotation resupply mission every 164 days.

3.5.1.3.2.2 Mass and Size Characteristics

Mass growth allowances of 33 percent were applied to base hardware; 20 percent was applied to CTV, CEM, and payload packaging hardware.

Based on the estimated equipment sizes stated in the Rockwell study, a total of 20 4.3m dia x 8.8m length (14.5 x 29 ft) EDM's plus the eight shelter modules will require transportation. The average mass (with growth) of these modules is approximately 6 800 kg (15,000 lbs). Since a lunar transport vehicle (LTV) is required to deliver the EDM's between lunar orbit and the lunar surface, this unit also requires delivery to Earth orbit.

Upon reaching Earth orbit, the EDM's and LTV's can be arranged in combinations for delivery to lunar orbit and subsequent delivery to the lunar surface. Transportation requirements between Earth orbit and the lunar surface are summarized in figure 3.5-7. As defined in the reference study, two LTV's are required to deliver equipment to the lunar surface so the first two base equipment delivery flights will include LTV's. A crew transfer vehicle (CTV) is included to provide emergency return capability for the crew should the LTV crew compartment be uninhabitable. As indicated in figure 3.5-7 options of either delivering two or four EDM are possible with the resulting mass being respectively 61 360 kg (135,000 lbs) and 88 640 kg (195,000 lbs).

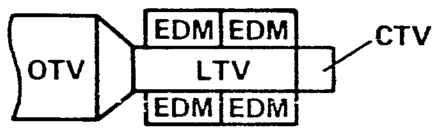
LTV transportation requirements from lunar orbit to the lunar surface will be 20 900 kg (46,000 lbs) or 35 450 kg (78,000 lbs) depending whether two or four EDM's are delivered. These masses include the crew/equipment module for the landing crew.

Subsequent OTV flights will be required to deliver the remainder of the EDM's. Again, an option of either two or four EDM's may be delivered per flight. In addition to the EDM's, a propulsion module (PM) is also included for replenishment of the LTV. After the third OTV/payload flight reaches lunar orbit, the LTV at the LSB will launch and rendezvous with the OTV. LTV

EARTH ORBIT – LUNAR ORBIT	LUNAR ORBIT – LUNAR SURFACE

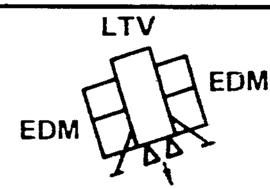
FIRST TWO FLIGHTS

Earth orbit – lunar orbit



	OTV P/L	KG	(LB)
DELIVER*	2 EDM	61 360	(135,000)
	4 EDM	88 640	(195,000)
RETURN	CTV	6 820	(15,000)

Lunar orbit – lunar surface



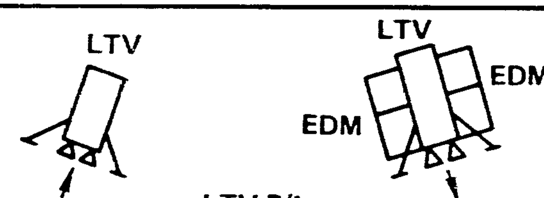
LTV P/L KG	(LB)
20 910	(46,000)
35 460	(78,000)

SUBSEQUENT FLIGHTS

Earth orbit – lunar orbit

	OTV P/L	KG	(LB)
DELIVER	2 EDM	46 820	(103,000)
	4 EDM	76 820	(169,000)
RETURN		0	0

Lunar orbit – lunar surface



LTV P/L

KG	(LB)	KG	(LB)
—		20 910	(46,000)
—		35 460	(78,000)
5909	(13,000)		

* PLUS CTV

Figure 3.5-7. LSB Surface Equipment Delivery

replenishment and payload transfer will be performed and the LTV returned to the LSB. Payload masses for the LTV are the same as for first two flights.

System elements required to perform the crew rotation/resupply functions include CTV for six crew, two RM containing a total 20 000 kg (44,000 lbs) of supplies and a PM for LTV propulsion replenishment. Individual masses for these elements are as follows:

Element	kg	lbs
CTV	7 270	16,000
RM(2)	21 820	48,000
PM	41 820	92,000

Arrangement and total masses for the crew rotation/resupply elements on their delivery flight to lunar orbit and to the lunar surface are shown in figure 3.5-8. Payload to lunar orbit is 70 910 kg (156,000 lbs) and 27 730 kg (61,000 lbs) to the lunar surface with the latter again including the crew/equipment module and two RM's.

3.5.1.3.2.3 Operational Constraints

Nominal operational constraints are similar to the OLS mission (para. 3.4.1.3.2.4). Nonpolar lunar parking orbits may be used, unless a polar base is planned. Low inclination orbits may allow shorter trip times for logistics flights. For a specific base latitude, a lunar orbit can be selected that will allow surface stay times of any designated duration. If the planned stay time t is overstayed, the next in-phase ascent opportunity generally occurs after 27.5 - t days (t in days).

Safety and Abort Constraints-Except for the additional criterion that lunar descents to a landing site in the vicinity of an occupied surface base shall be piloted by a minimum flight crew of two, safety criteria for the lunar-surface base mission are the same as for the OLS mission.

If the base is not supported by an OLS, either Earth-based rescue capability or onsite abort capability (see below) capable of evacuating the base crew to Earth is required. The standard logistics system may be used but must be held in a standby capacity in orbit (unless it can be launched by a single launch) so that the next available lunar departure opportunity may be used. Departure opportunities occur about every 10 days. The maximum 10-day wait for a departure opportunity plus 3 days for lunar transfer and 1 day for lunar-orbit operations equal a maximum wait time for rescue of 14 days. The best available lunar orbit will be used to avoid wait time in lunar orbit prior to descent.

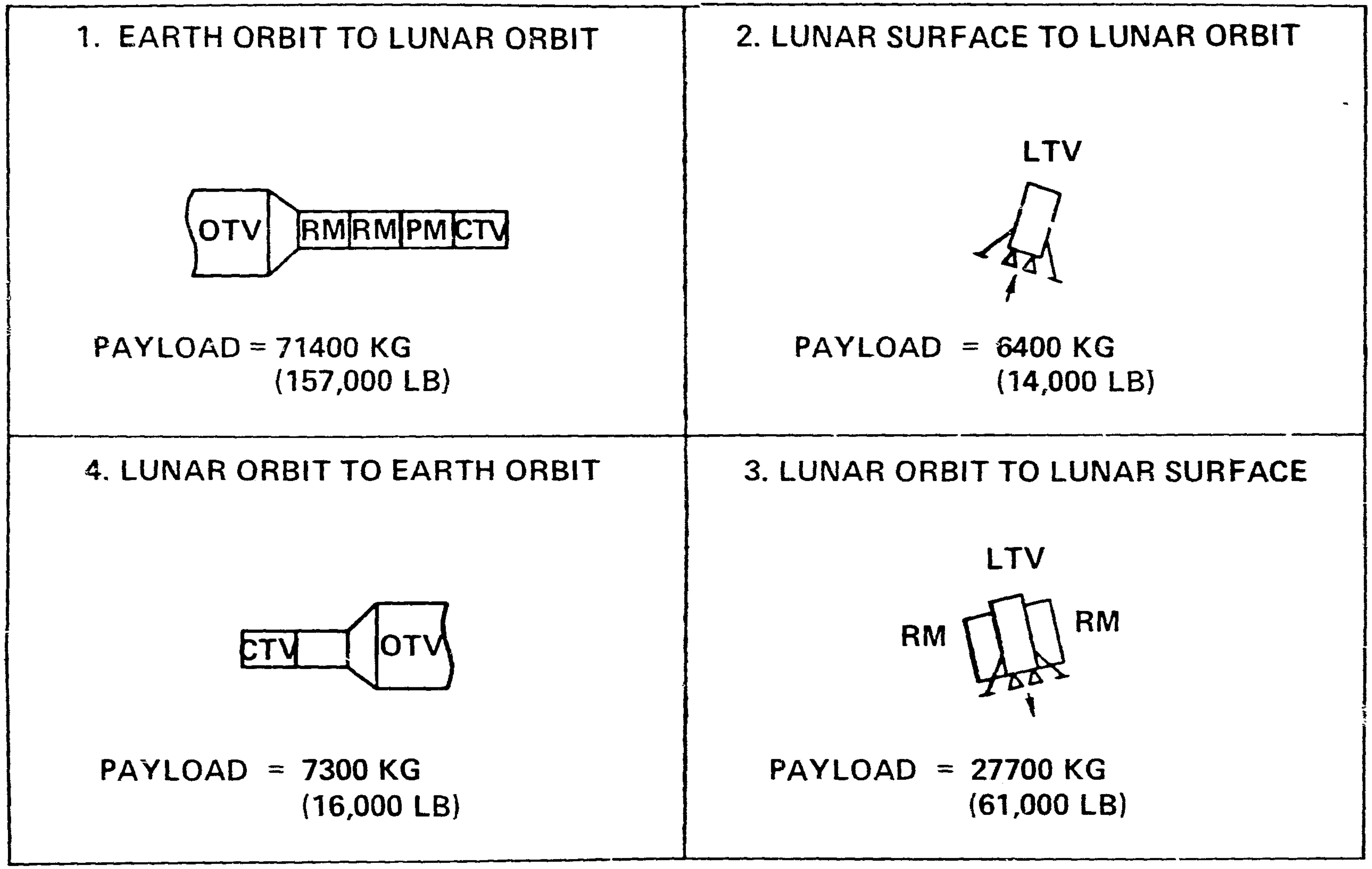


Figure 3.5-8. LSB Crew Rotation/Resupply Delivery

The onsite abort alternative provides one or more abort vehicles, similar to the lunar transport vehicle, fully fueled on the surface at the base. This vehicle would land in the normal way and be refueled on the surface by a subsequent cargo flight. If abandoning the base is necessary, the abort vehicle(s) ascend to a lunar orbit of opportunity suitable for immediate trans-Earth injection and must have sufficient total delta V capability to enter a low Earth orbit.

Earth-based rescue was assumed in sizing transportation system.

3.5.1.4 Mission/Transportation Modes and Operations

3.5.1.4.1 Transportation Options

Transportation systems are required for Earth launch, delivery to lunar orbit using OTV's and delivery to the lunar surface with LTV's.

The principal OTV and LTV transportation candidates are shown in table 3.5-11.

Table 3.5-11. LSB OTV and LTV Transportation Candidates

Transportation System	OTV	LTV
LO_2/LH_2 single stage	x	x
LO_2/LH_2 1-½ stage	x	x
LO_2/LH_2 common stage	x	
LO_2/LH_2 two stage		x
LO_2/MMH single stage		x
LO_2/MMH common stage	x	

The principal Earth launch vehicle candidates are:

- Space Shuttle (SS)
- Heavy Lift Launch Vehicle (HLLV)

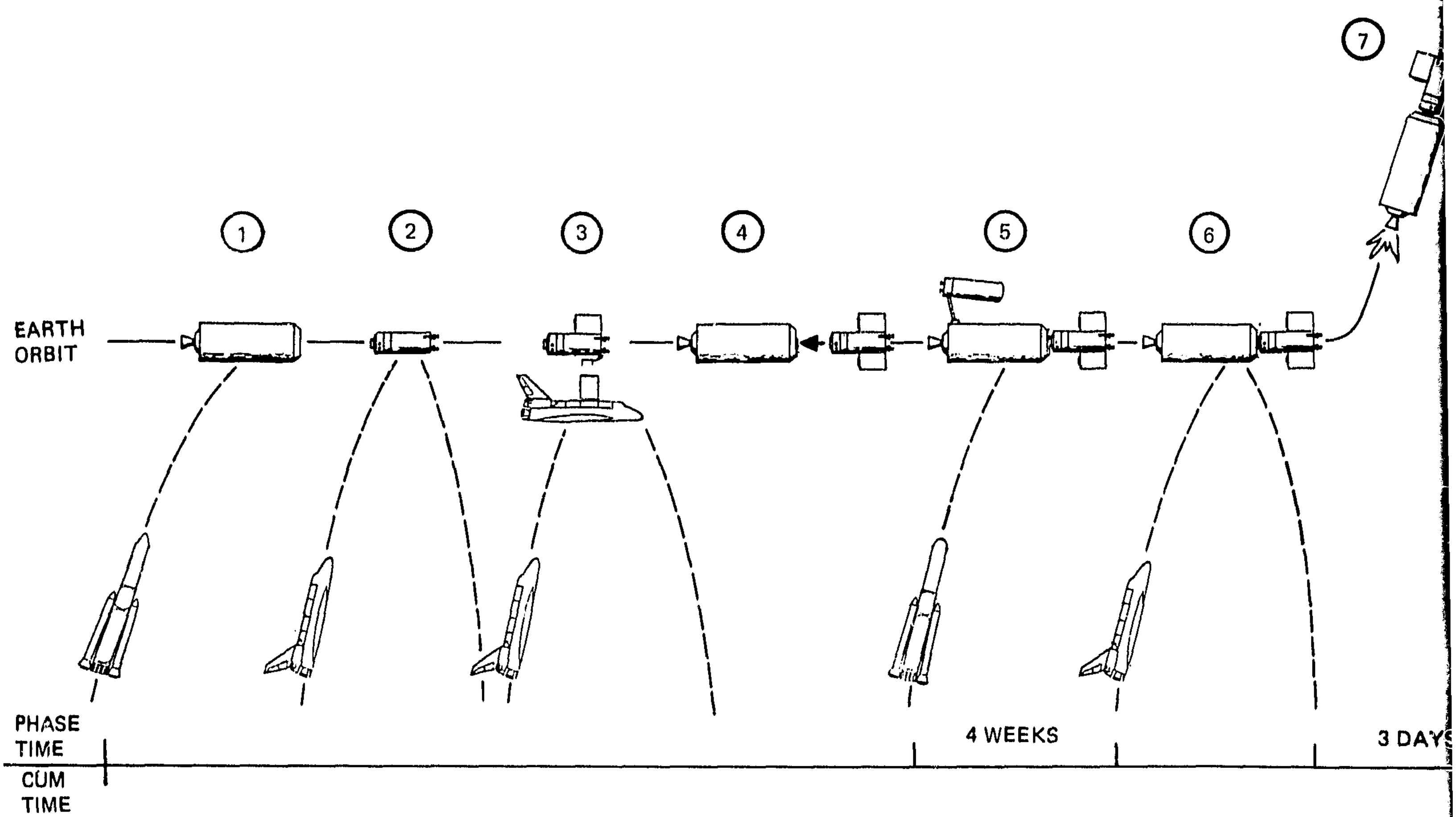
3.5.1.4.2 Representative Transportation Mode and System

The representative sequence employs the space shuttle and a HLLV as Earth launch systems, and single-stage LO_2/LH_2 OTV and LTV's.

3.5.1.4.2.1 Transportation Sequence

Typical sequences and operations associated with the LSB mission are depicted in figure 3.5-9. Transportation systems employed include both space shuttle and HLLV launch vehicles, single stage OTV and single stage LTV. The principal transportation features associated with these missions are as follows:

SURFACE EQUIPMENT DELIVERY OPERATI
(FIRST TWO FLIGHTS)



1	2	3	4	5	6	7
• DELIVER OTV TO ORBIT WITH HLV	• DELIVER LTV TO ORBIT WITH SS	• DELIVER EQUIPMENT MODULES TO ORBIT WITH SS • DOCK EQUIPMENT MODULES TO LTV	• DOCK LTV TO OTV	• DELIVER OTV TANKERS WITH HLV (2 LAUNCHES) • FUEL OTV • TANKER DISPOSAL	• DELIVER FLIGHT CREW • TOP OFF OTV AND LTV PROPELLANTS	• TRANSLU INJECTIO • COAST • LUNAR O INSERTIO

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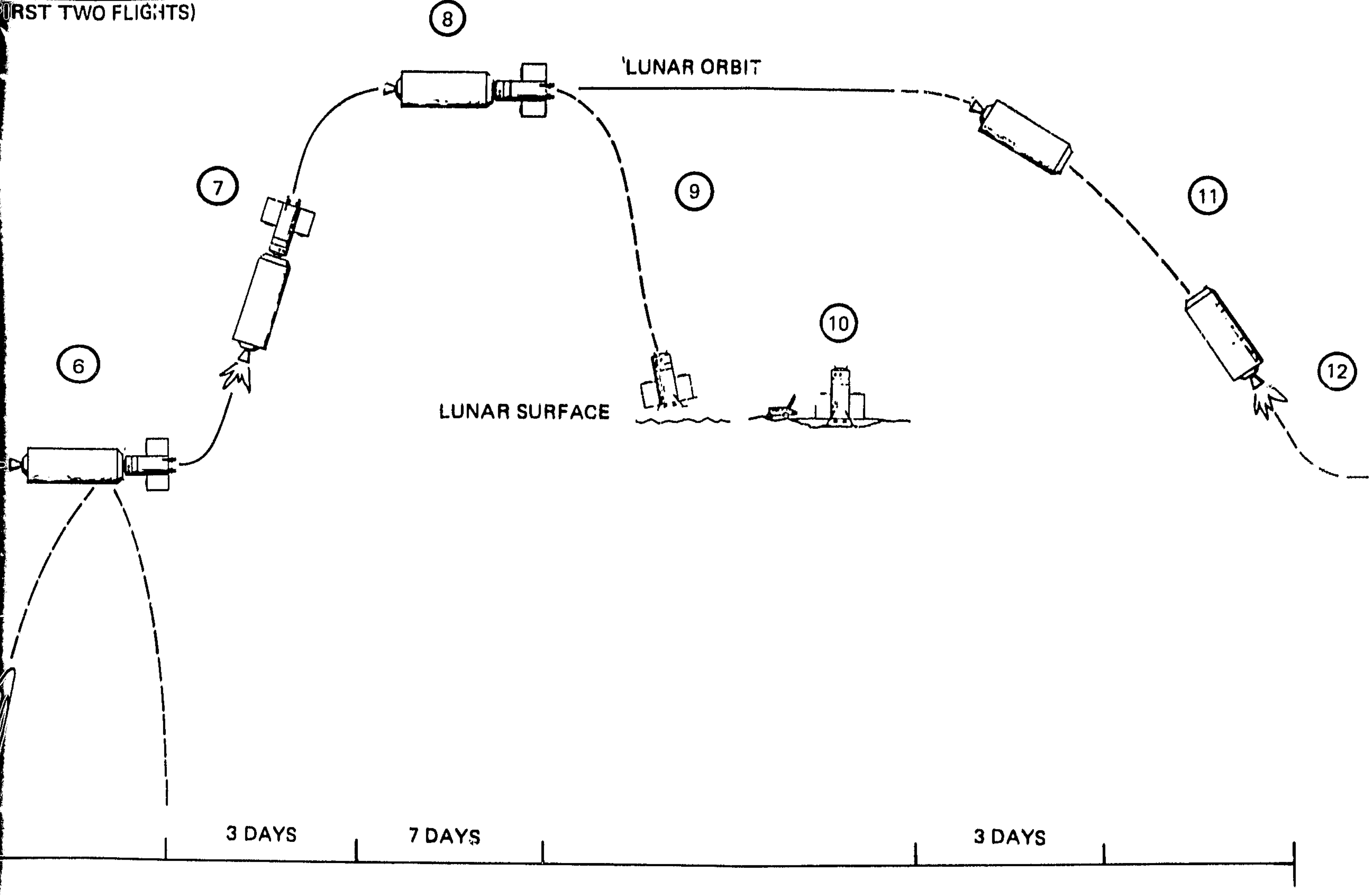
/

IPMENT DELIVERY OPERATIONS
IRST TWO FLIGHTS)



6	7	8	9	10	11	12
DELIVER FLIGHT CREW TOP OFF OTV AND LTV PROPELLANTS	• TRANSLUNAR INJECTION • COAST • LUNAR ORBIT INSERTION	• LUNAR ORBIT CIRCULARIZATION • WAIT FOR LANDING SITE ALIGNMENT	• LTV DESCENT AND LANDING	• OFF-LOAD EQUIPMENT • PREPARE BASE SITE • ASSEMBLE BASE	• OTV PERFORMS TRANSEARTH INJECTION • COAST • OTV PERFORMS EARTH ORBIT INSERTION	• DELIVER 2ND LTV AND EQUIP MODULES • REPEAT STEPS 2 THRU 11

Figure 3.5-9 LSB Mission Transportation Sequence (Sheet 1)

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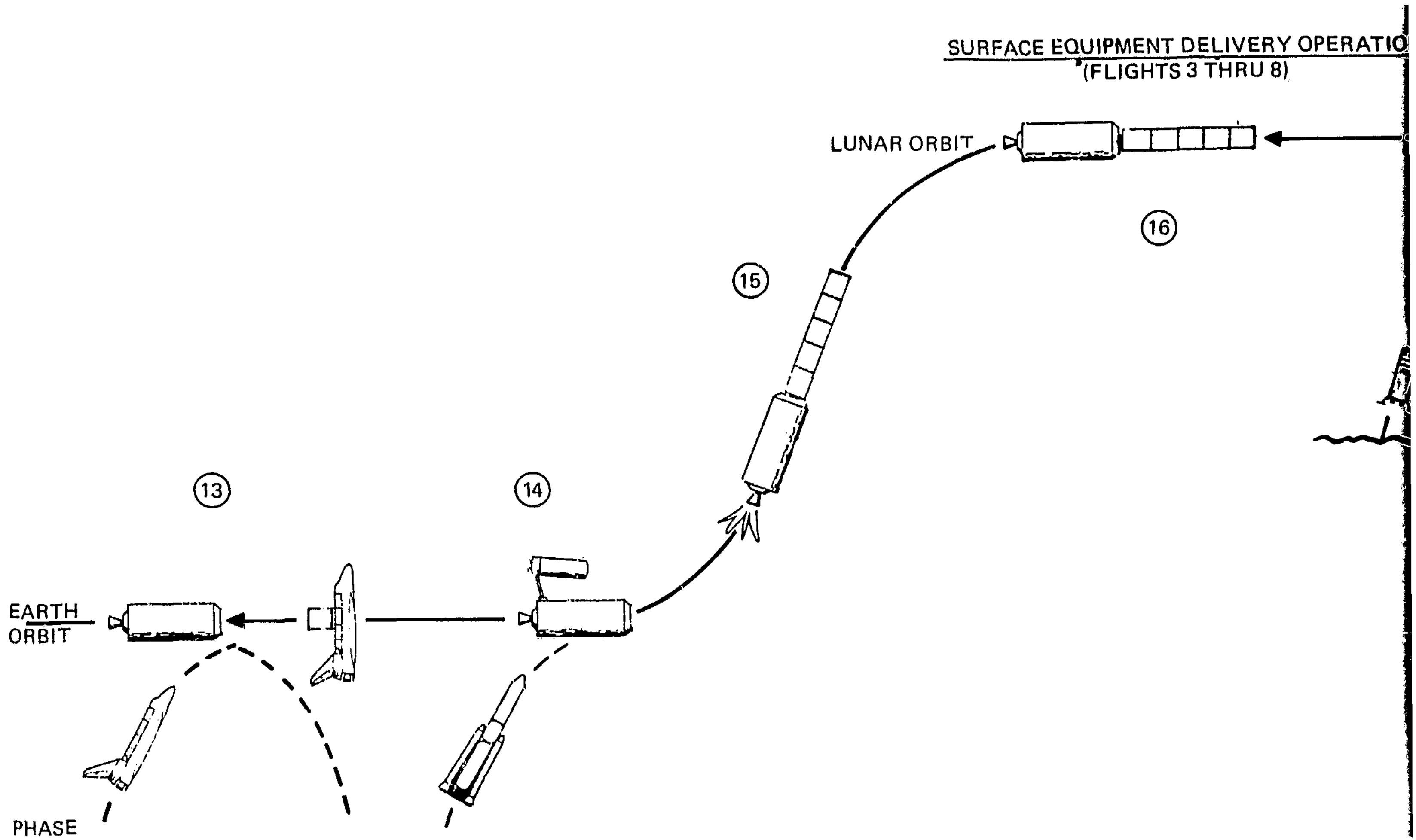

	13	14	15	16
PHASE TIME	3 WEEKS	4 WEEKS	3 DAYS	7 DAYS
CUM TIME				
	• DELIVER EQUIPMENT MODULES AND LTV PROPELLANT MODULE WITH SS	• DELIVER OTV TANKERS WITH HLV (2 LAUNCHES) • FUEL OTV • TANKER DISPOSAL	• TRANS LUNAR INJECTION • COAST • LUNAR ORBIT INSERTION	• WAIT FOR LANDING SITE ALIGNMENT

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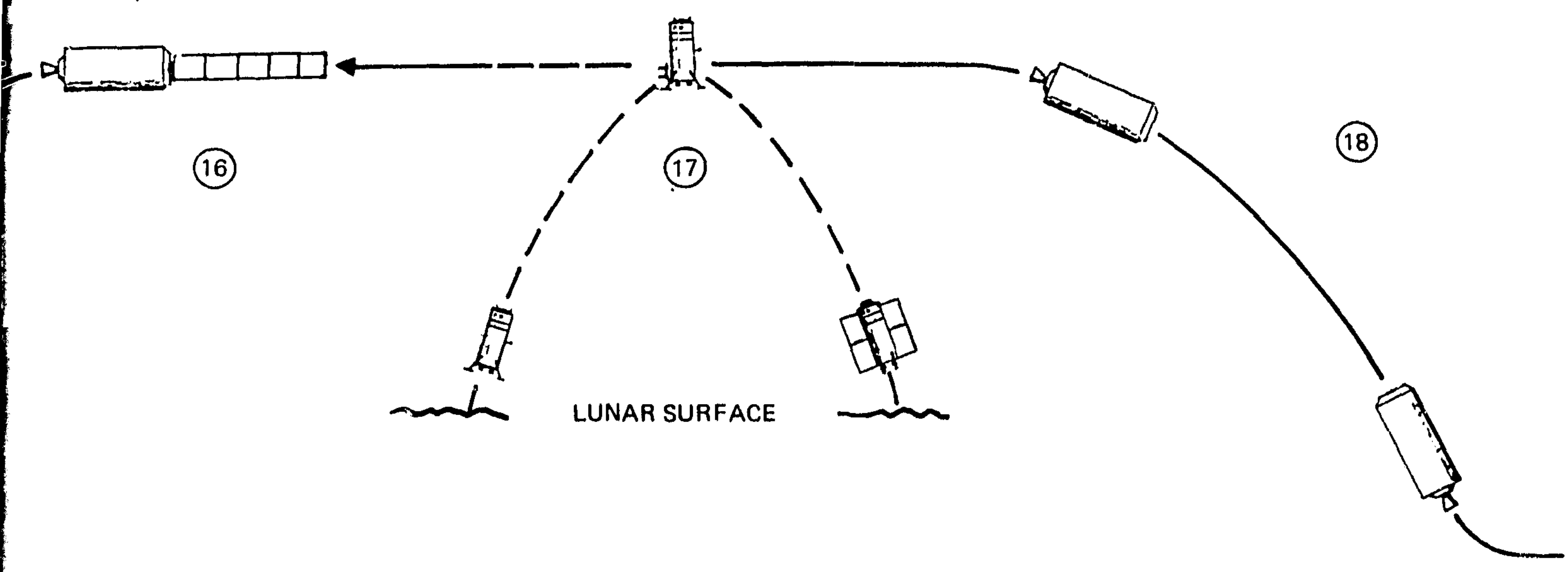
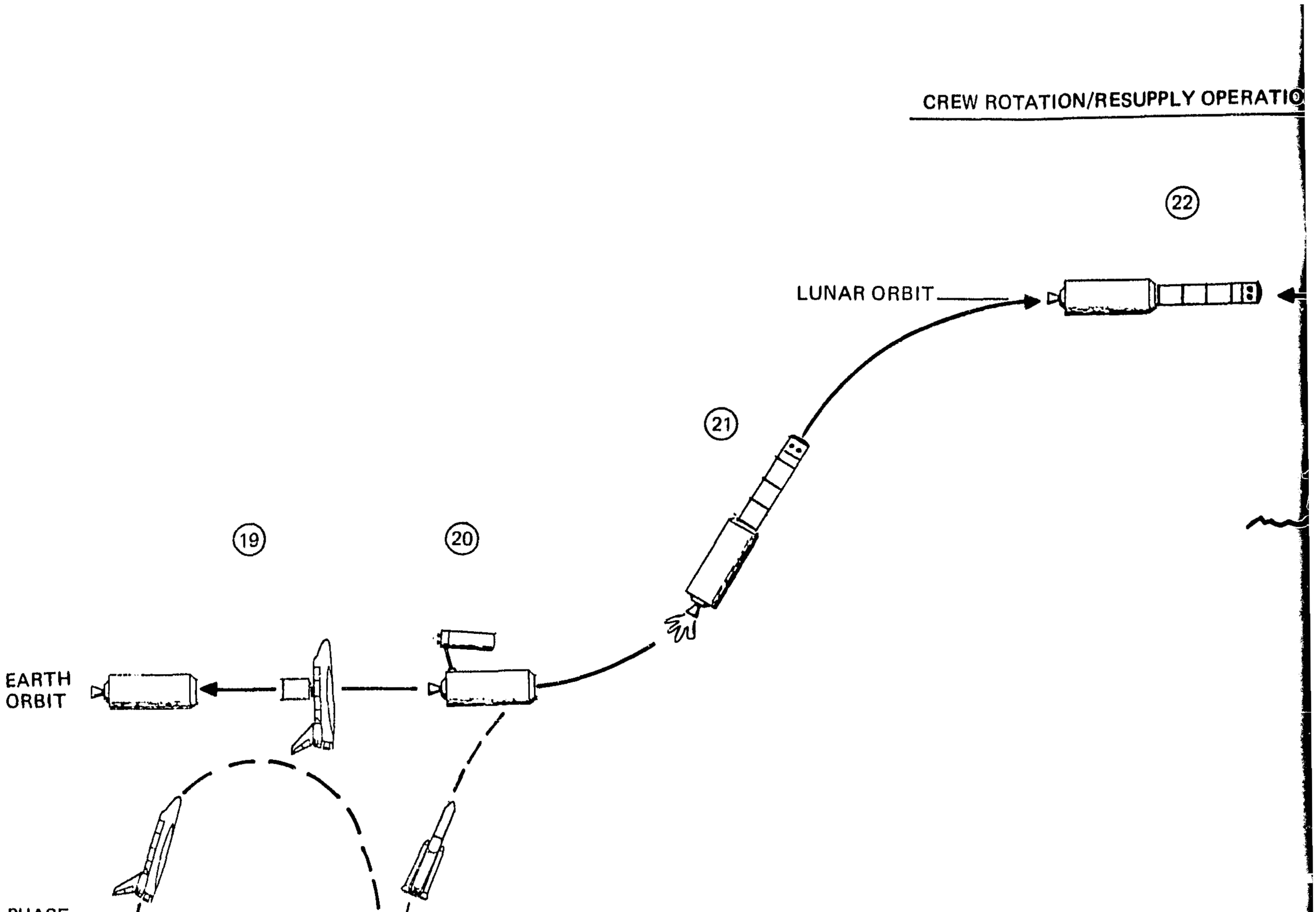
RFACE EQUIPMENT DELIVERY OPERATIONS
(FLIGHTS 3 THRU 8)



7 DAYS	3 DAYS	3 DAYS
16	17	18
• WAIT FOR LANDING SITE ALIGNMENT	• LTV ASCENT/ RENDEZVOUS WITH OTV • LTV DOCKS WITH OTV AND RECEIVES EQUIPMENT MODULES • LTV PROPULSION REPLENISHMENT • LTV DESCENT	• OTV PERFORMS TRANS EARTH INJECTION • COAST • OTV PERFORMS EARTH ORBIT INJECTION

Figure 3.5-9 LSB Mission Transportation Sequence (Sheet 2)



	19	20	21	22
PHASE TIME	2 WEEKS	4 WEEKS	3 DAYS	7 DAYS
CUM TIME				
	• DELIVER PAYLOADS WITH SS • RESUPPLY MODULES • LTV PROP MODULES • CREW TRANSFER VEHICLE • LSB CREW REPLACEMENT	• DELIVER OTV TANKERS WITH HLV (2 LAUNCHES) • FUEL OTV • TANKER DISPOSAL	• TRANS LUNAR INJECTION • COAST • LUNAR ORBIT INSERTION	• WAIT FOR LUNAR SURFACE SITE ALIGNMENT

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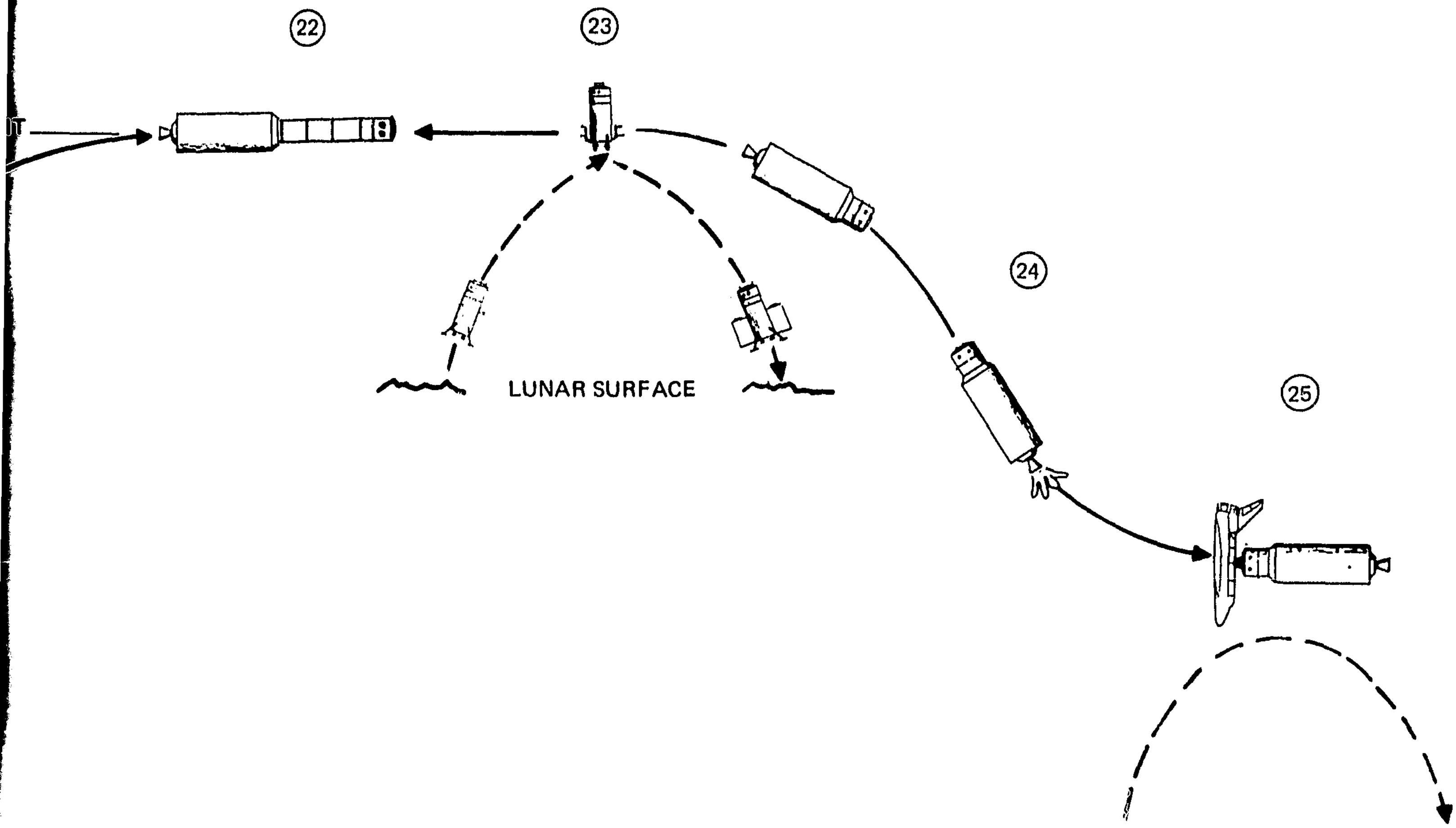
CREW ROTATION/RESUPPLY OPERATIONS



22	23	24	25
7 DAYS	3 DAYS	3 DAYS	1 DAY
• WAIT FOR LUNAR SURFACE SITE ALIGNMENT	• LTV ASCENT WITH 1/2 OF BASE CREW • RENDEZVOUS AND DOCK WITH OTV/CARGO • LTV RECEIVES RESUPPLY MODULE • LTV PROPULSION REPLENISHMENT • CREW EXCHANGE	• OTV PERFORMS TRANS EARTH INJECTION • OTV PERFORMS EARTH ORBIT INJECTION	• LSB CREW PICKUP BY ORBITER • REPEAT STEPS 19 THRU 25 EVERY 165 DAYS

Figure 3.5-9 LSB Mission Transportation Sequence (Sheet 3)

FOLDOUT FRAME

- A large single stage OTV is delivered to Earth orbit with a HLLV.
- EDM and LTV are delivered to Earth orbit using SS and are docked to the OTV.
- Fueling of the OTV and LTV is completed using a tanker launched by a HLLV.
- The OTV delivers these payloads to lunar orbit where the LTV transports the EDM's to the lunar surface.
- The OTV is returned to Earth orbit with the operations associated with delivery of another LTV and EDM's being repeated.
- The next six OTV flights will deliver EDM's as well as PM's to lunar orbit.
- When the OTV reaches lunar orbit, a manned LTV will launch from the lunar surface and perform rendezvous and docking operations followed by delivery of the cargo back to the lunar surface.
- Crew rotation/resupply flights make use of the SS to deliver the elements to Earth orbit, OTV for delivery to lunar orbit and a LTV for transportation to the surface.
- Return of crews to Earth orbit will be followed by retrieval using the space shuttle.

A mission history indicating key transportation events, elasped time, ΔV and mass remaining is presented in table 3.5-12.

3.5.1.4.2.2 Transportation Sizing

Since the majority of the flights during the course of the LSB program will be for crew rotation/resupply, the OTV should be sized for this flight. As indicated on the performance map of figure 3.5-10, an OTV sized in this manner will also satisfy all other delivery requirements including that of transporting four EDM's per flight (after the first two flights).

A performance map for the LTV is shown in figure 3.5-11. The LTV size was selected for crew rotation/resupply; it delivers 2 EDM's per trip for base buildup.

3.5.1.4.2.3 Operational Factors

Mission Profiles and Rendezvous Techniques—The LSB buildup and support operation can probably be carried out without a dedicated Earth orbit support facility. The single-stage OTV is then not constrained to return to a particular Earth orbit, but can return to any low-inclination (e.g., 30

Table 3.5-12. LSB Mission History — Single Stage LO_2/LH_2 OTV

Event	Elapsed Time	ΔV		Mass Remaining	
	Hr	MPS	FPS	KG	LB
Initial mass	0	—	—	355,100	782,900
Translunar injection		3,345	10,974	168,600	371,700
Lunar orbit insertion		968	3,175	136,000	299,900
Separate payload 71,400 kg (157,400 lb)				64,600	142,500
Return payload 7,300 kg (16,000 lb)		968	3,175	71,900	158,500
Transearth injection				57,900	127,600
Earth orbit insertion		3,198	10,491	28,300	62,500
(OTV inert)				(21,000)	(46,500)
(Payload)				(7,300)	(16,000)

Table 3.5-12. (Continued) LSB Mission History – Single Stage LO_2/LH_2 LTV

Event	Elapsed Time	Δ V		Mass Remaining	
	Hr	MPS	FPS	KG	LB
Initial mass				70,700	156,000
Descent	2	2,194	7,198	42,400	93,500
Offload landed payload 21,300 kg (47,000 lb)	10			21,100	46,500
Ascent	676	2,020	6,627	13,200	29,100
(LTV inert)				6,800	(15,000)
(Ascent payload)				(6,400)	(14,100)

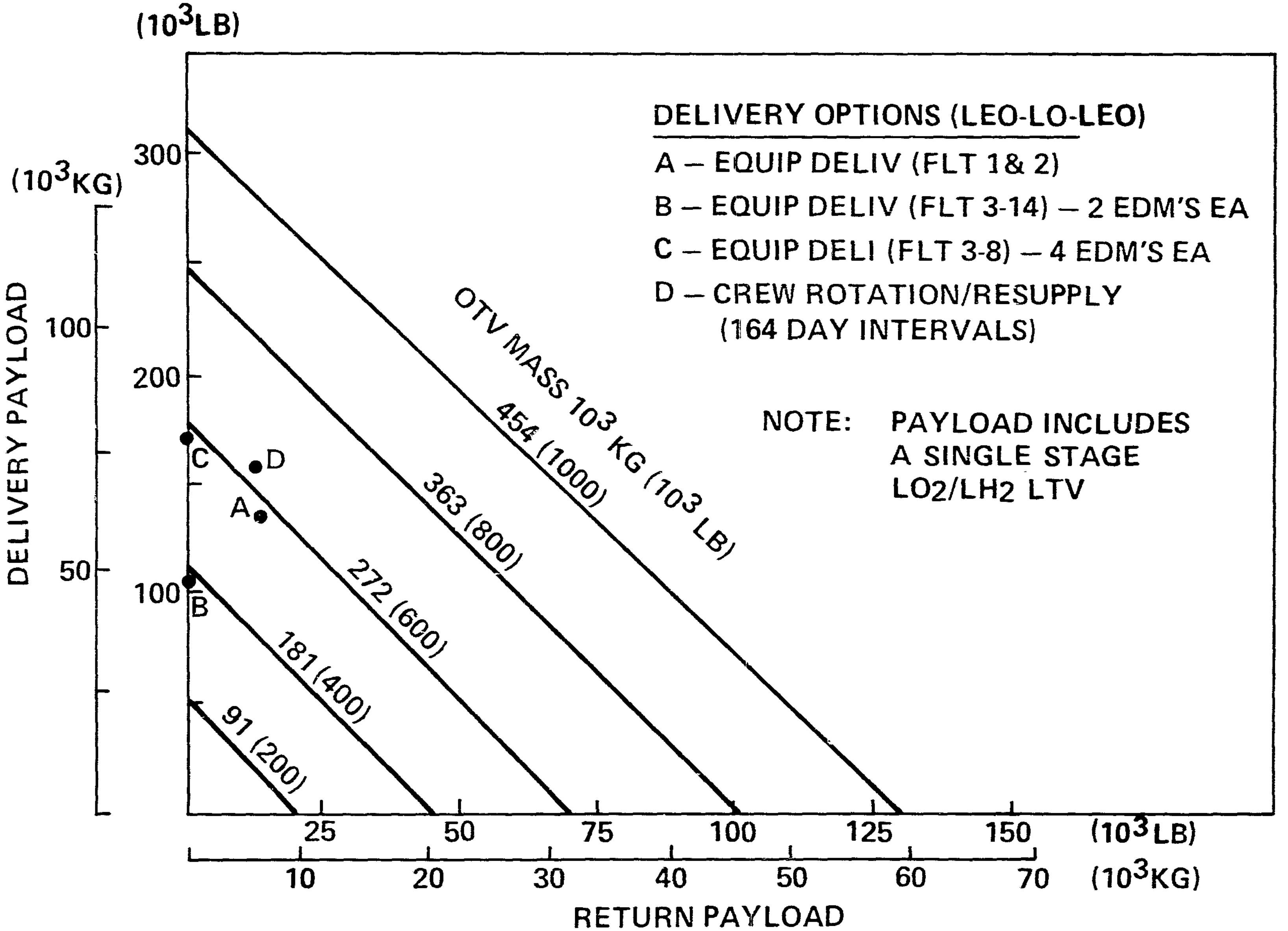


Figure 3.5-10. Single Stage LO_2/LH_2 OTV Capability for LSB

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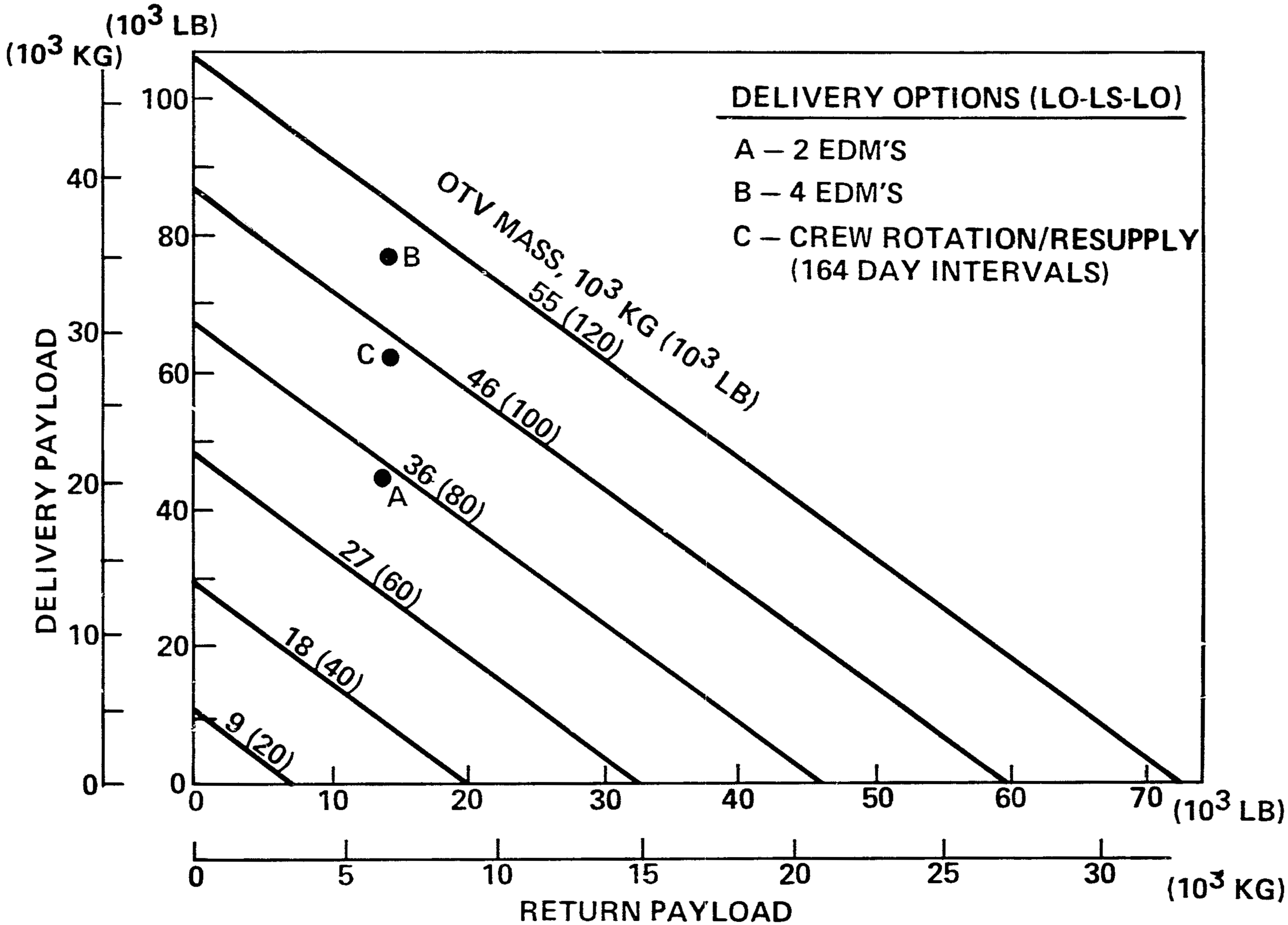

Figure 3.5-11. Single Stage LO_2/LH_2 LTV Capability for LSB

degrees) orbit. Lunar orbits may be selected for the particular base landing site, as was true for Apollo.

Modest plane changes, required for departure from low-inclination lunar orbits at times other than the in-plane opportunity, are acceptable. Logistics flight round trips of short duration, i.e., 8-10 days, are therefore feasible as shown in figure 3.5-12.

If missions are staged from a facility in a particular Earth orbit, the 55-day lunar flight opportunity repeat cycle as described for the OLS will be a constraint also for the lunar surface base. Resulting timelines are also shown in figure 3.5-12.

Rendezvous techniques will be conventional concentric sequences as was true for the OLS. In this case, rendezvous at the moon will be LTV active (OTV passive) and at Earth, Earth launch vehicle active (OTV again passive).

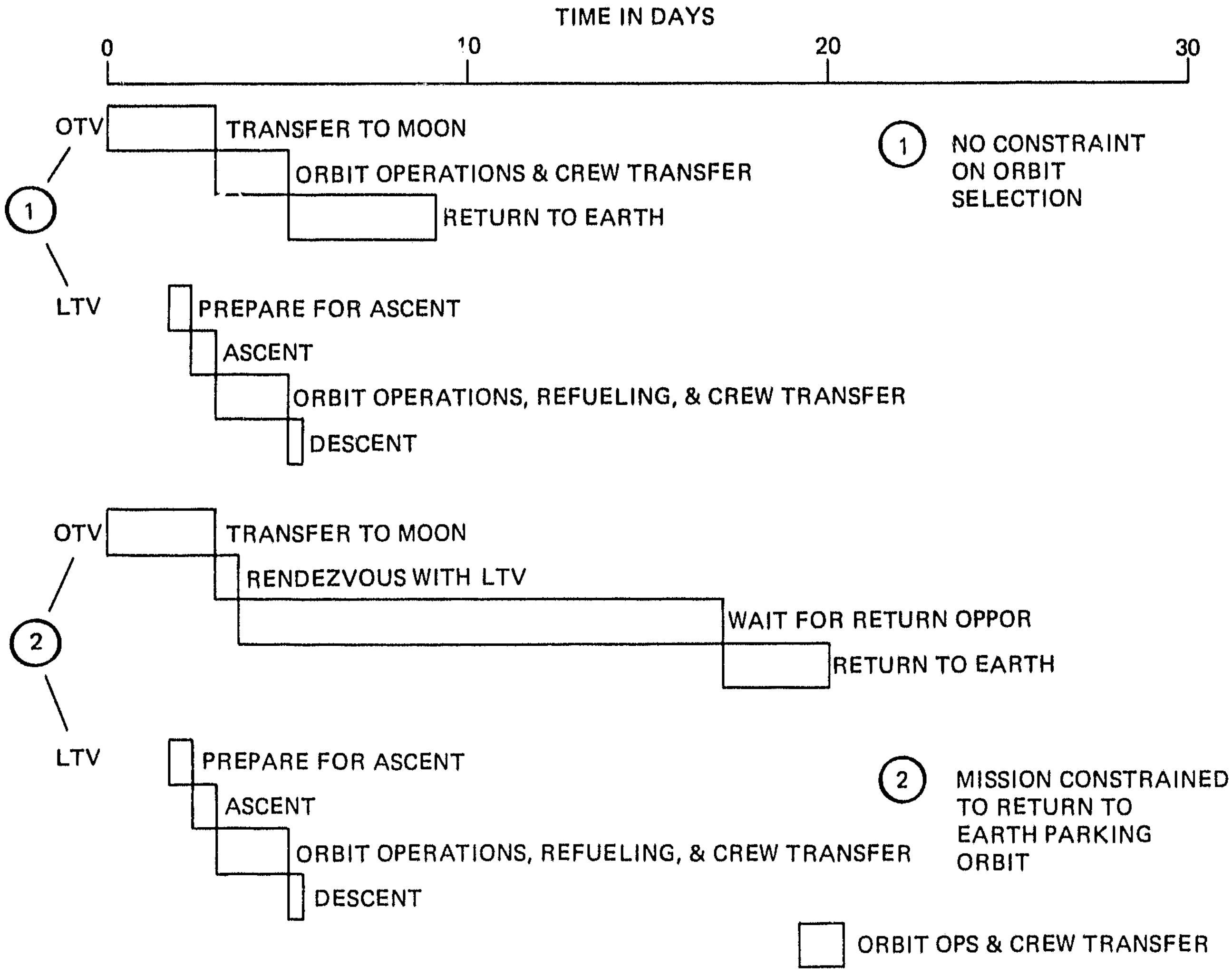

Figure 3.5-12. Typical Earth-Lunar Flight Timelines

Crew Involvement--Operations crews and mission crews will be involved in essentially all phases of the lunar surface base.

Crew involvement simplifies transportation operations such as rendezvous, docking, orbital assembly, lunar landing, and cargo handling on the lunar surface where the crew can be relied upon as the principal control element. Crew timelines (e.g., time required to transfer from one vehicle to another) do not have any identified significant effect on transportation requirements. The crew will assemble the base from the transported hardware. The reference study estimated 27 lunar manmonths would be required.

The nominal crewman stay time at the 12-man surface base is two logistics intervals, i.e., 328 days. The nominal stay time for the 6-month base is 164 days (3 times the 55-day translunar opportunity repeat cycle). If an OLS is employed as an aid to mission operations, an additional wait time at the OLS may be required typically 14 to 30 days for Earth return opportunity.

Control Functions and Requirements--Control functions and requirements are the same as for the OLS (para. 3.4.1.4.2.3) with the added requirement that lunar landings must be precision targeted to a designated landing site within 100 meters (300 feet) or less. A homing beacon on the lunar surface can be used as a landing aid.

Network Support--Network support will be similar to that for the OLS (para. 3.4.1.4.2.3). The LSB buildup and resupply transportation operations will be largely autonomous. Principal data return means will be hard copy, tapes, and samples.

3.5.1.4.2.4 Earth Launch Requirements Summary

Delivery of the eight shelter modules and 20 EDM's will require eight OTV flights assuming four EDM's are delivered on each flight after the first two flights. A total of two OTV flights per year will also be required for crew rotation/resupply.

Sixteen shuttle flights are required to deliver base hardware to Earth orbit; 8 shuttle flights and 18 HLLV flights are required to support transportation operations for base buildup. Four shuttle flights and 6 HLLV flights annually are required to support crew rotation and resupply.

3.5.1.4.3 Transportation Options Comparison and Evaluation

3.5.1.4.3.1 Size and Performance Comparison

Parametric performance maps were developed for all of the OTV candidates except the 1-1/2 stage system which was on a point design basis. Delivery payloads for the first two OTV flights,

subsequent OTV flights delivering either two or four EDM's and crew rotation/resupply requirements are super-imposed on the performance maps of figure 3.5-13 and 3.5-14.

OTV mass comparison for all candidates is presented in figure 3.5-15 for the crew rotation/resupply flight. The range of OTV masses is approximately 227 300 kg (500,000 lbs) for the LO_2/LH_2 1-1/2 and common stages and 318 200 kg (700,000 lbs) for the LO_2/MMH common stage. The smallest full stage is again the return stage of the 1-1/2 stage system at 25 000 kg (55,000 lbs).

Performance maps were also developed for the LTV transportation candidates with the exception of the 1-1/2 stage concept which was analyzed on a point design basis. These performance plots are shown in figures 3.5-16 and 3.5-17. Payload for this delivery is the crew equipment module and the two resupply modules.

Mass comparison of the LTV transportation concepts is presented in figure 3.5.18. Mass of the LO_2/MMH single stage is approximately 50 000 kg (110,000 lb) which is 6 820 kg (15,000 lbs) greater than the LO_2/LH_2 single stage. However, due to its greater density the LO_2/MMH system provides a total LSV length which is 3m (10 ft) shorter.

3.5.1.4.3.2 Earth Launch Requirements Comparison

The number of Earth launches required to deliver OTV hardware and fuel necessary to deliver the LSB surface equipment necessary to initiate the mission is shown in figure 3.5-19. Several of the OTV candidates may be launched using the space shuttle since they are dimensionally compatible. All of the OTV concepts can be launched with the HLLV with a considerable reduction in number of Earth launches. Space shuttle flights are shown with one of the HLLV options since only a portion of the HLLV capability would be required to complete the delivery of the OTV systems.

In general, for those OTV candidates that can use either launch vehicle, use of the HLLV results in only one-fourth as many Earth launches. Earth launches required to deliver OTV hardware and fuel necessary for the LSB annual requirements is shown in figure 3.5-20. Again, use of the HLLV requires only one-third to one-fourth as many launches as required when using the space shuttle.

3.5.1.4.3.3 Operational Comparison

The operational comparison for the OLS program (para. 3.4.1.4.3.3) applies with the following additional consideration:

The LSB operation is not necessarily tied to the 55-day logistics repeat cycle as was the OLS because

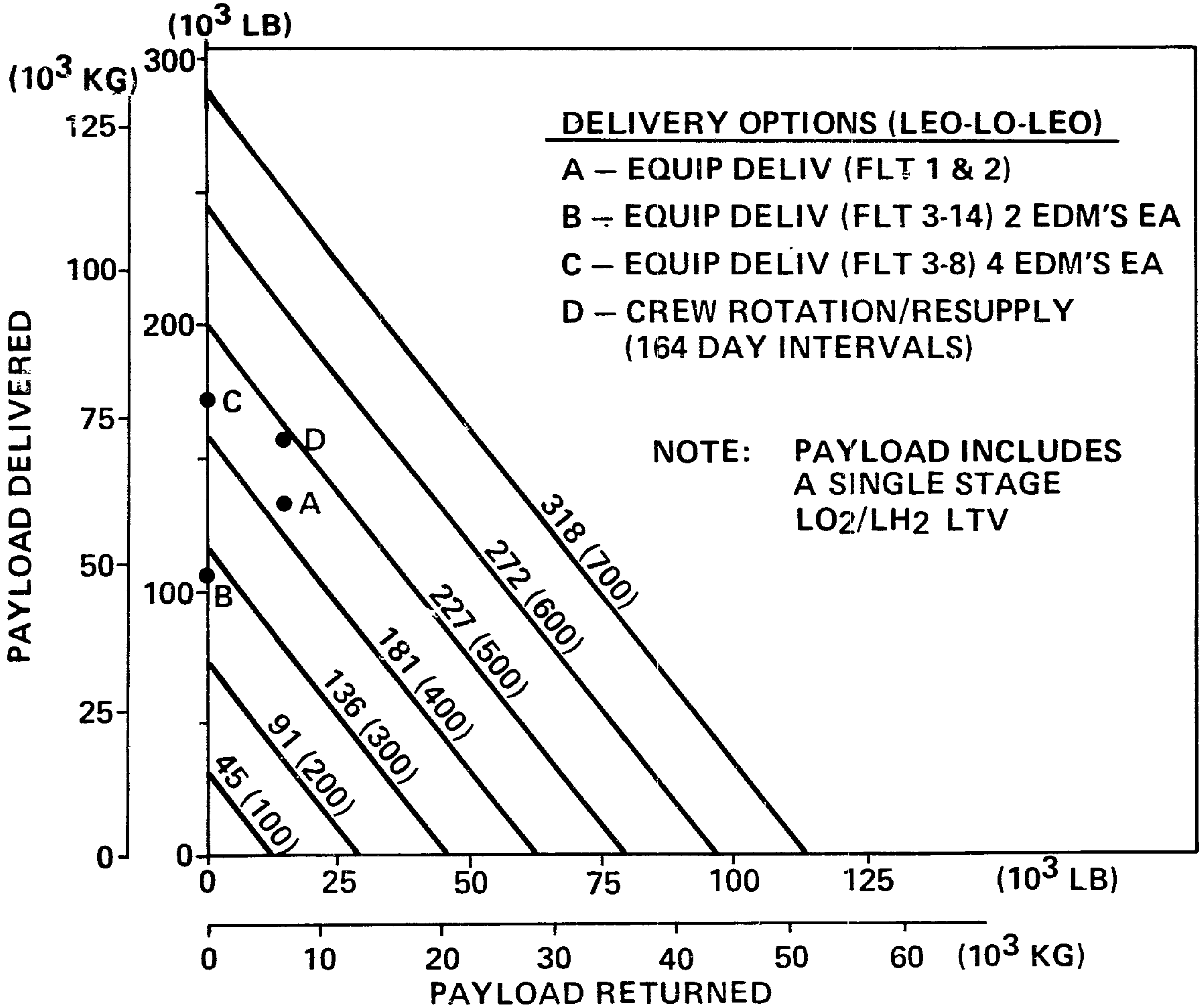

Figure 3.5-13. Common Stage LO_2/LH_2 OTV Capability for LSB

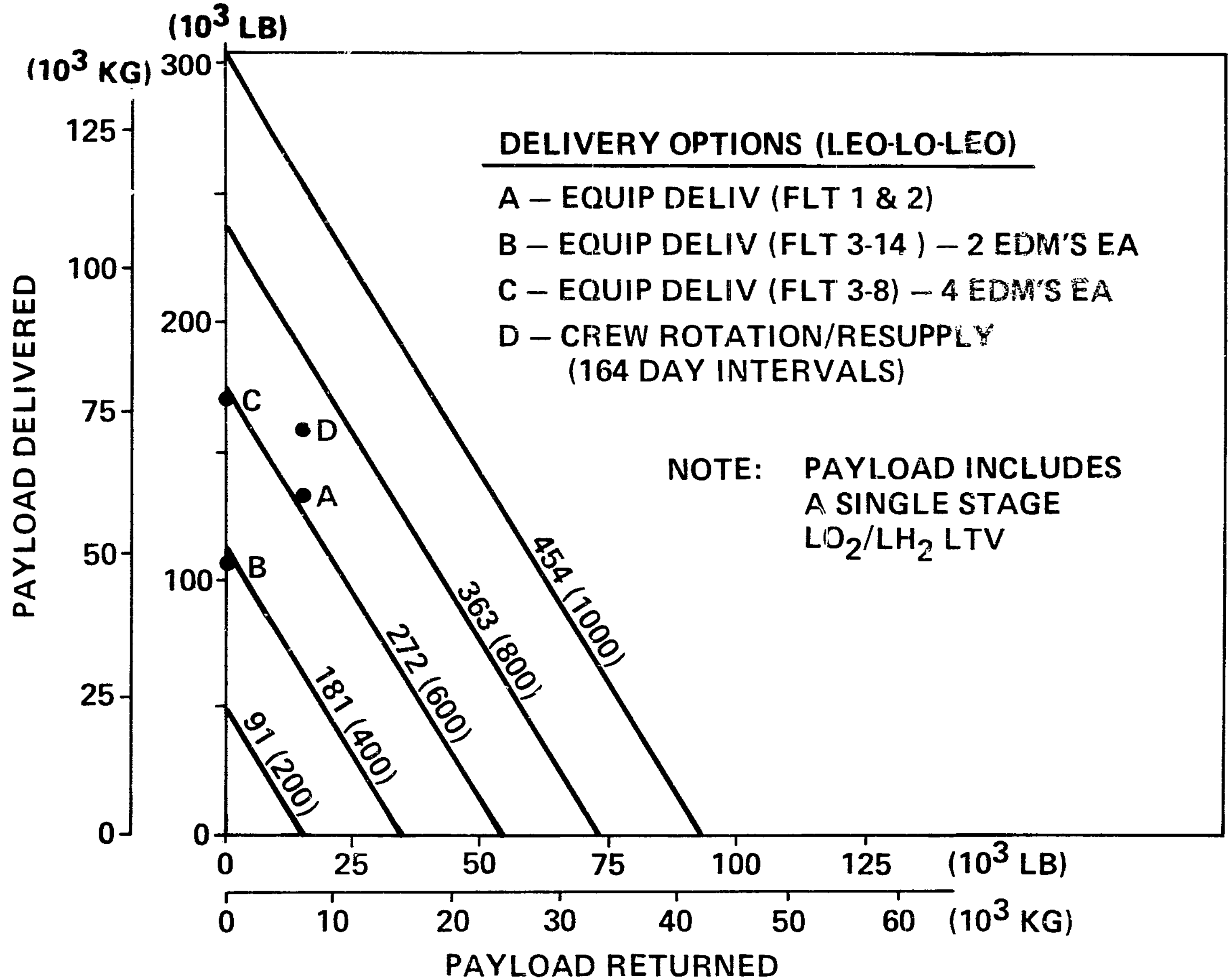

Figure 3.5-14. Common Stage LO_2/MMH OTV Capability for LSB

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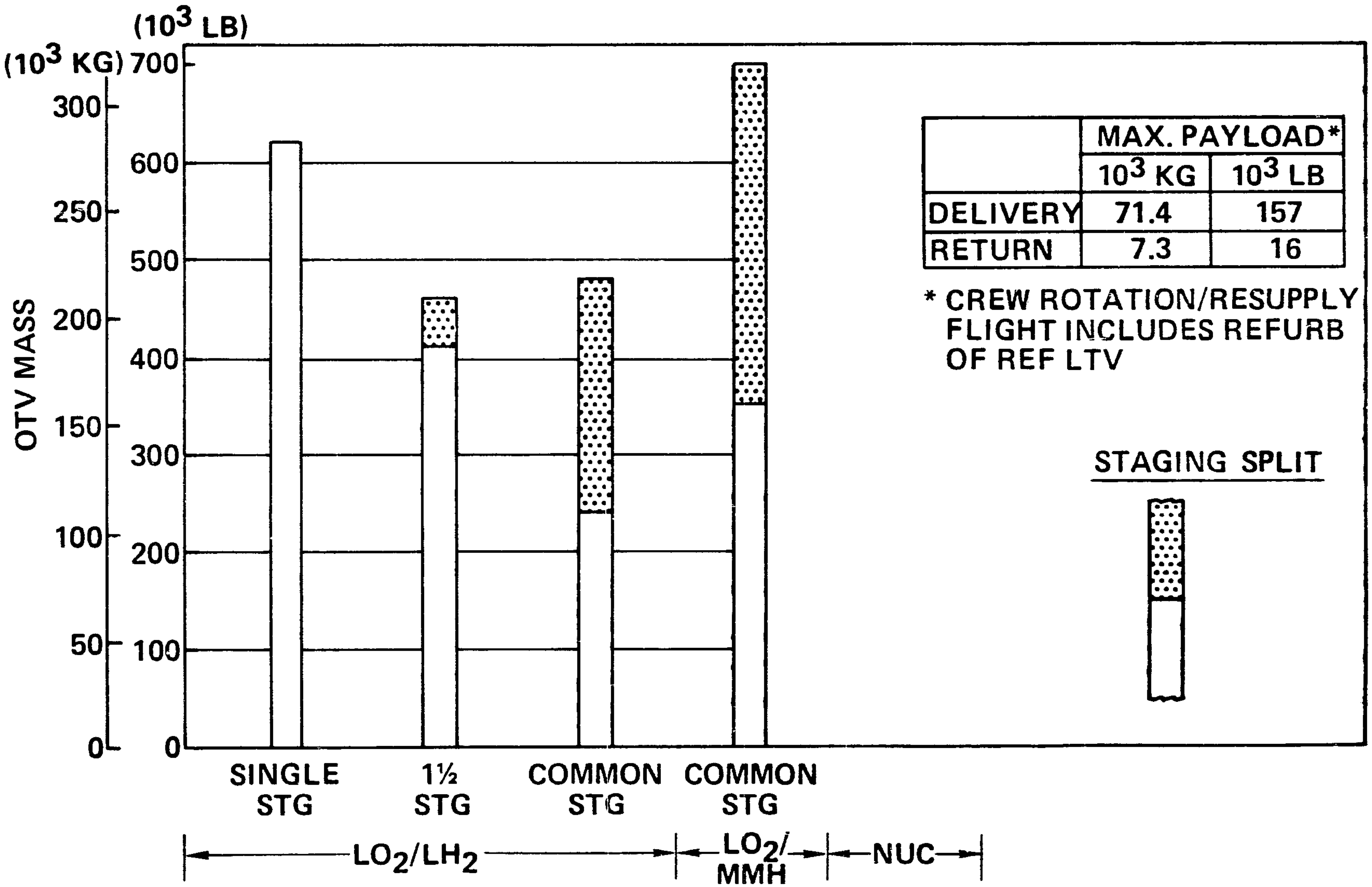

	MAX. PAYLOAD*	
	10^3 KG	10^3 LB
DELIVERY	71.4	157
RETURN	7.3	16

* CREW ROTATION/RESUPPLY FLIGHT INCLUDES REFURB OF REF LTV

Figure 3.5-15. OTV Comparison for LSB

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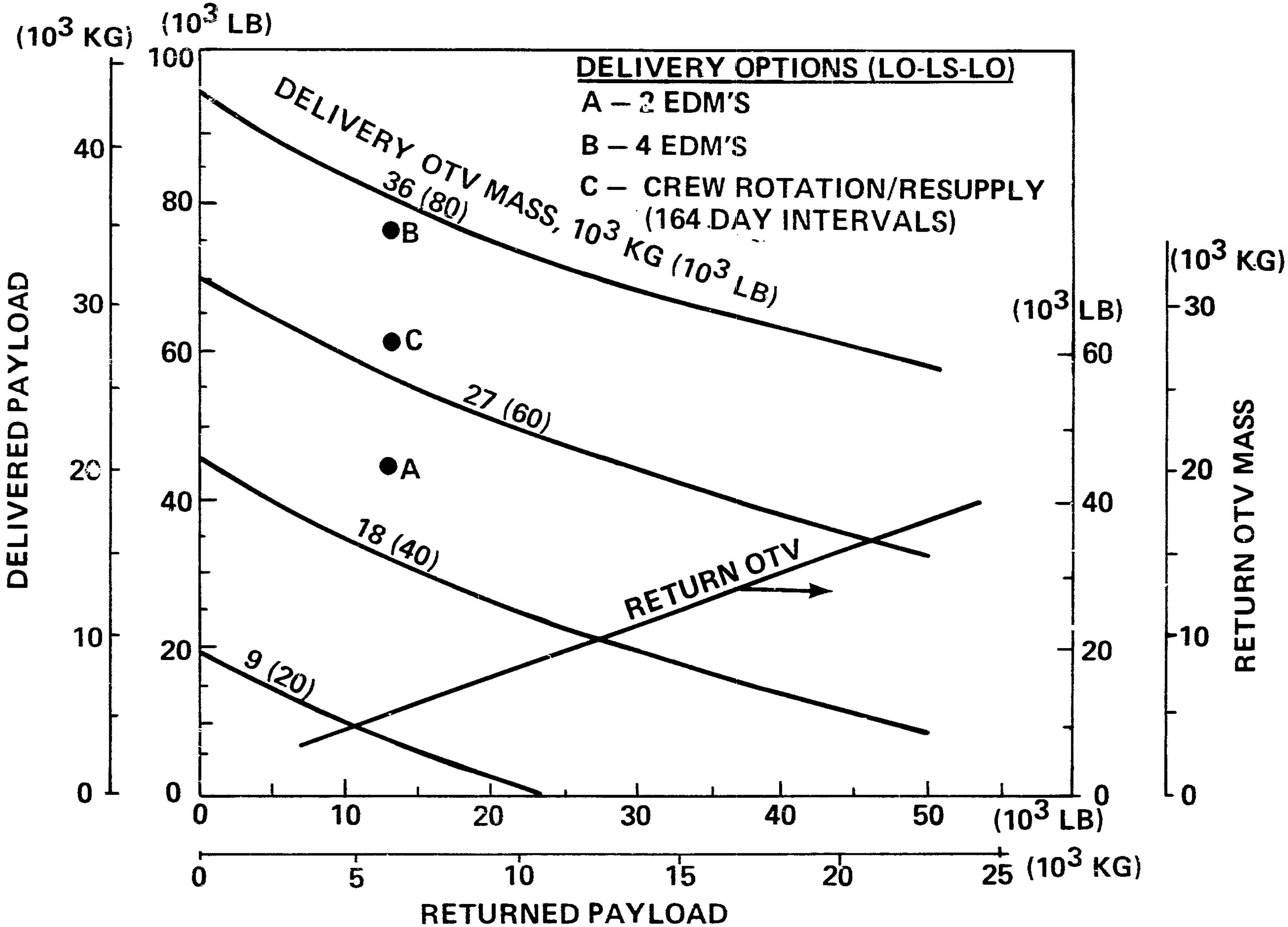


Figure 3.5-16. Two Stage LO_2/LH_2 LTV Capability for LSB

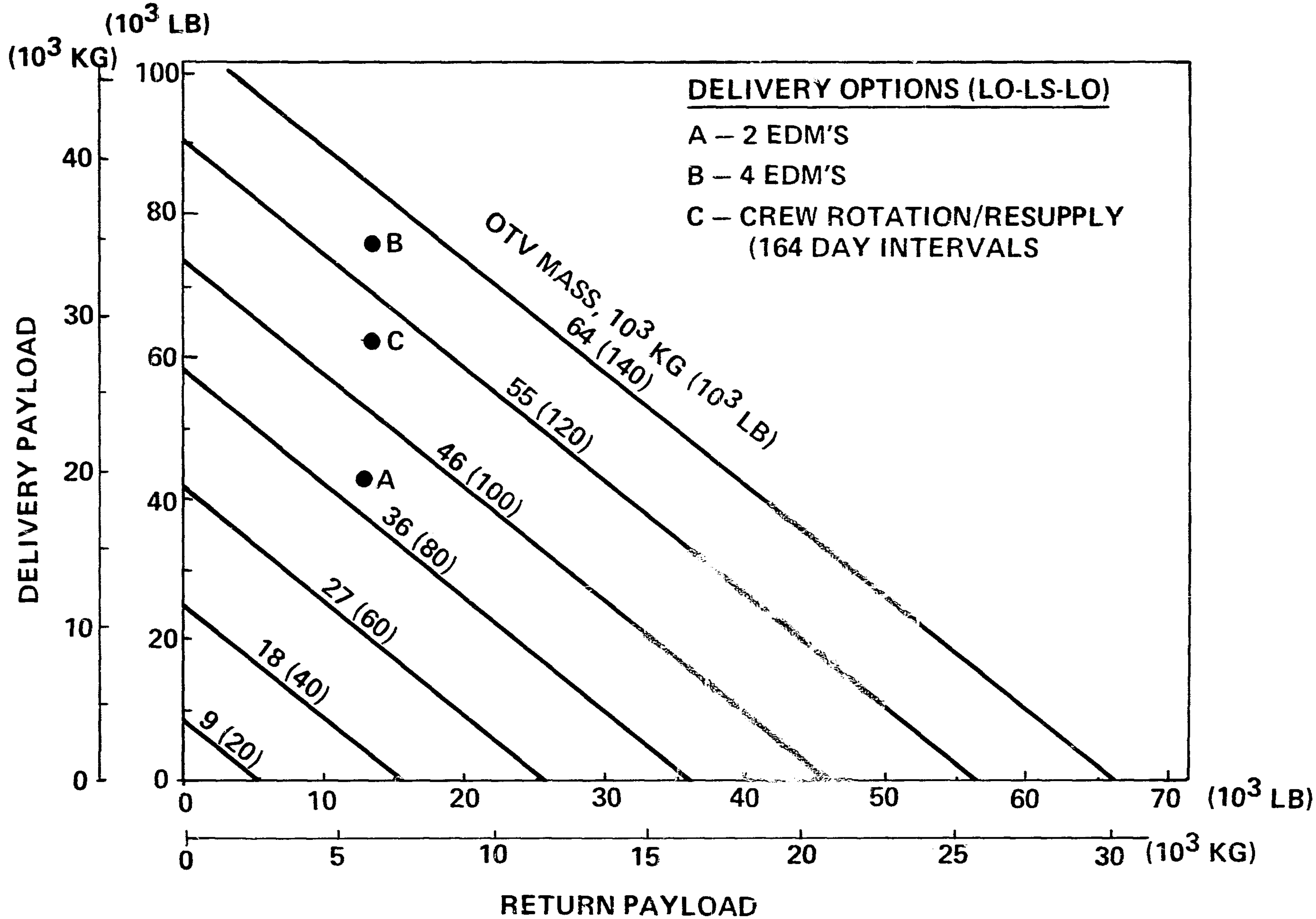
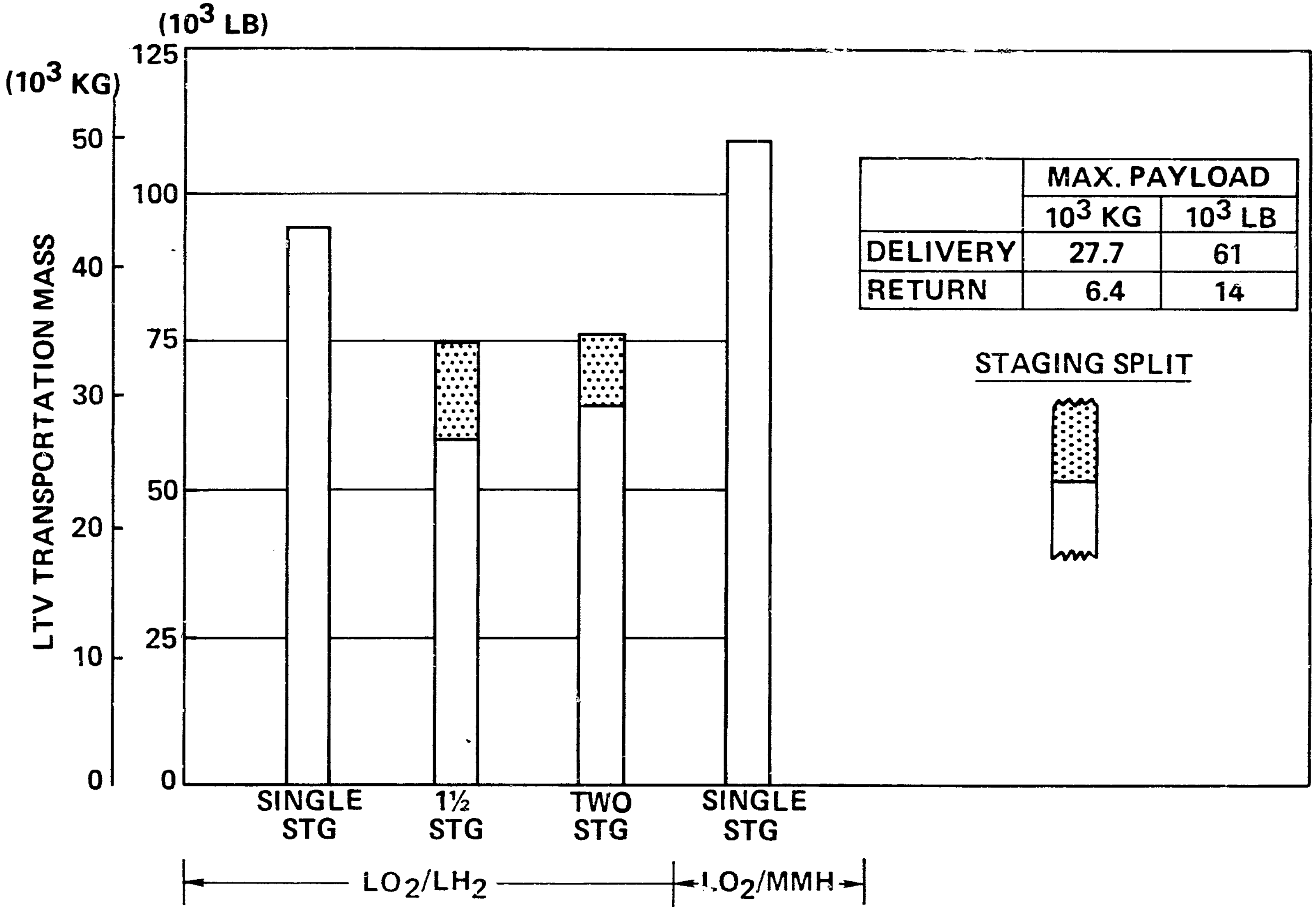


Figure 3.5-17. Single Stage LO_2/MMH LTV Capability for LSB



	MAX. PAYLOAD	
	10^3 KG	10^3 LB
DELIVERY	27.7	61
RETURN	6.4	14

Figure 3.5-18. LTV Comparison for LSB

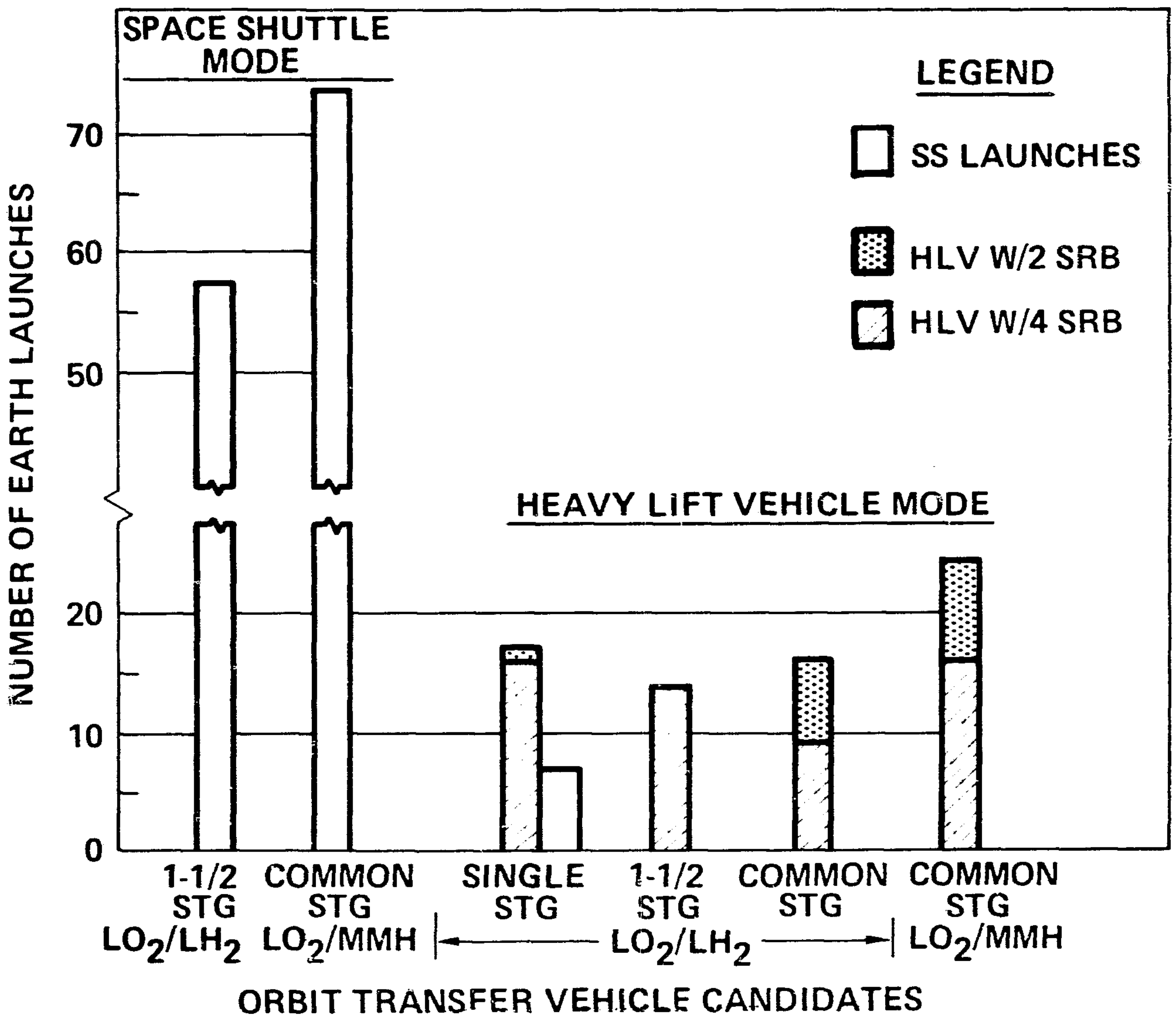


Figure 3.5-19. Earth Launches Required for LSB OTV System (Mission Start-Up)

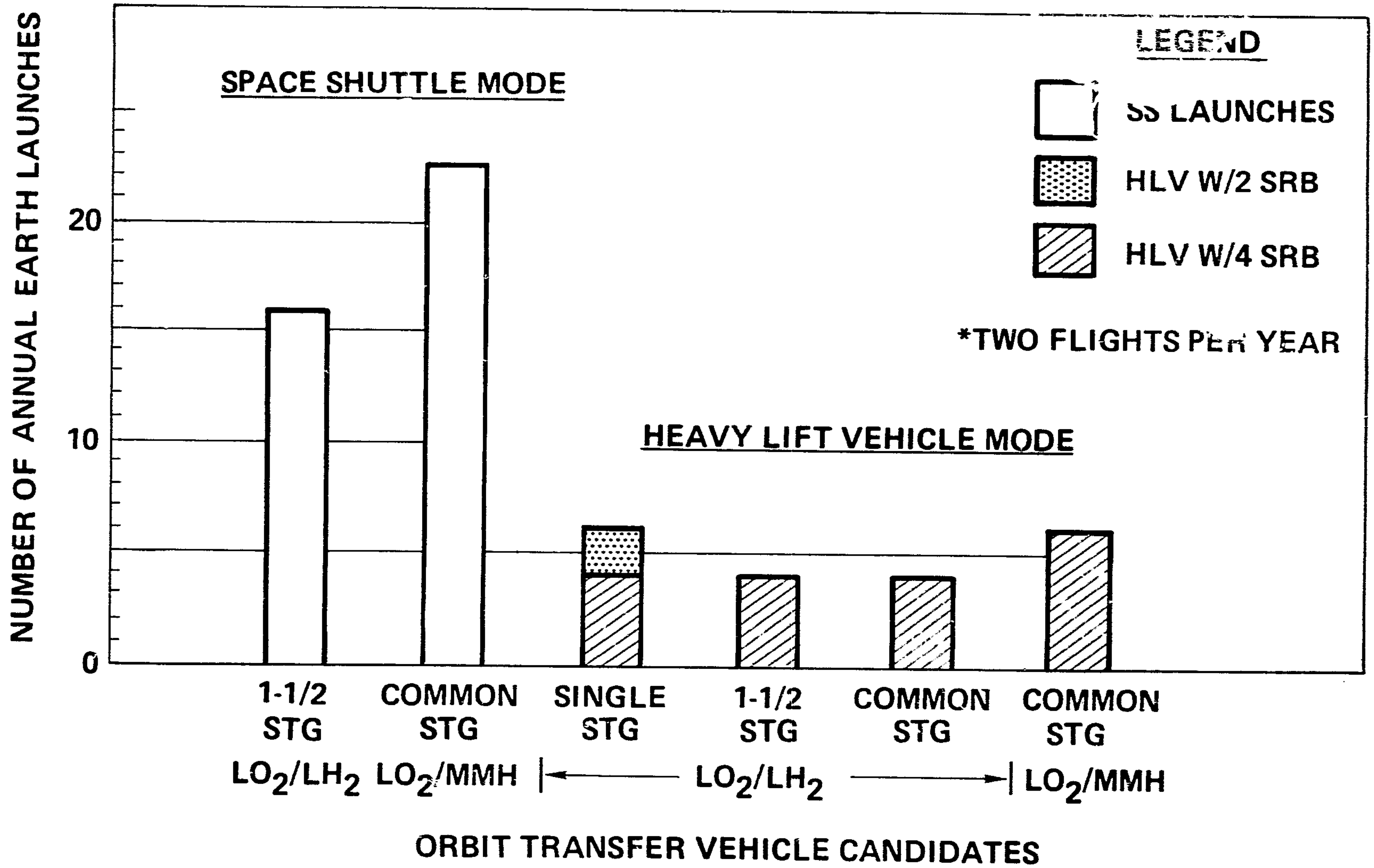


*Figure 3.5-20 Earth Launches Required for LSB OTV System (Annual Crew Rotation/Resupply) **

1) there is no lunar orbiting facility, hence a particular (i.e., polar) lunar orbit need not be used;

2) the base elements will not be assembled into the operational configuration in Earth orbit as was the OLS, hence OTV return to a particular Earth orbit is not required.

However, if a common-staged LO_2/LH_2 OTV is used return to a particular Earth orbit is required since the two OTV stages must get together again for reuse. Since the LO_2/MMH common stages can be returned to Earth by the shuttle the restriction does not apply to the LO_2/MMH system.

3.5.1.4.3.4 Practicality Assessment

All modes investigated are judged to be practical, except that the 2-stage LO_2/LH_2 LTV has a center of gravity problem as was noted in para. 3.4.1.4.3.4. Pro's and con's for the OTV modes are summarized in table 3.5-13.

Table 3.5-13. OTV Pro's and Con's

		ADVANTAGES					DISADVANTAGES		
		COMPATIBLE WITH SHUTTLE LAUNCH AND RECOVERY	NO STAGE-TO-STAGE DOCKING	FULLY REUSABLE	ONLY ONE STAGE TO DEVELOP	NO LH_2	MULTIPLE DOCKING TO ASSEMBLE	TWO ACTIVE VEHICLES TO TRACK	CONSTRAINED TO RETURN TO PARTICULAR EARTH ORBIT
LO_2/LH_2	SINGLE STAGE		x	x	x				
	1½ STAGE	x					x		
	COMMON STAGE			x	x			x	x
LO_2/MMH COMMON STAGE		x		x	x	x		x	

3.5.2 SIX-MAN, SIX-MONTH BASE

A brief study of a six-man, six-month base was conducted. The base and crew could be delivered to the moon by two OTV/LTV flights using the transportation system selected for the 12-man semipermanent base. A third flight would pick up the crew at the end of the six-month surface mission. No resupply was required. The mission did not impose any unique transportation requirements, and could be adapted to the transportation systems described for the ILSS, OLS, or LSB missions.

3.6 MANNED PLANETARY EXPLORATION PROGRAM

Manned Mars landing is considered the representative mission for the manned planetary exploration program. The objectives of this mission are to conduct an in-depth science program dealing with: Mars planetology, effects of modifying forces, composition, environment and possible life forms.

The traditional mission vehicle identified by many studies has been an assembly of spacecraft modules propelled by a multistage nuclear propulsion system as illustrated in figure 3.6-1.

A major reference for this mission is the Boeing IMISCD study of 1968.

3.6.1 MANNED MARS LANDING MISSION

3.6.1.1 Mission Summary

3.6.1.1.1 General Description

The major system elements associated with a manned landing on Mars include a mission module (MM), Mars excursion module (MEM) and Earth entry module (EEM). These elements along with unmanned probes form the mission spacecraft as shown in figure 3.6-2.

The mission module crew compartment provides the six man crew with a shirt-sleeve environment, quarters for living functions, operations center, experiment laboratories, radiation shelter and many of the subsystems required to support the above functions. This compartment is occupied by crewmen for the entire mission, except during the time when three crewmen descend to the Martian surface and during the Earth entry phase of the mission.

The MEM is used to land a three man crew on the surface, provide crew quarters and operations center for 30 days and return the crew to the mission spacecraft.

The Earth entry module (EEM) configuration is a six-man blunted biconic. The EEM systems are designed for 1 day's occupancy prior to Earth entry. The heat shield is designed by the highest Earth entry velocity expected from the opposition mission. The unmanned probes are used to checkout the potential landing sites for the MEM, collect Mars orbital science data and explore the moons orbiting Mars.

3.6.1.1.2 Mission Assumptions and Constraints

Nominal mission assumptions and constraints are tabulated in table 3.6-1. These are adopted from the Boeing IMISCD study of 1968 with the addition of a small surface rover in the class of the Apollo LRV.

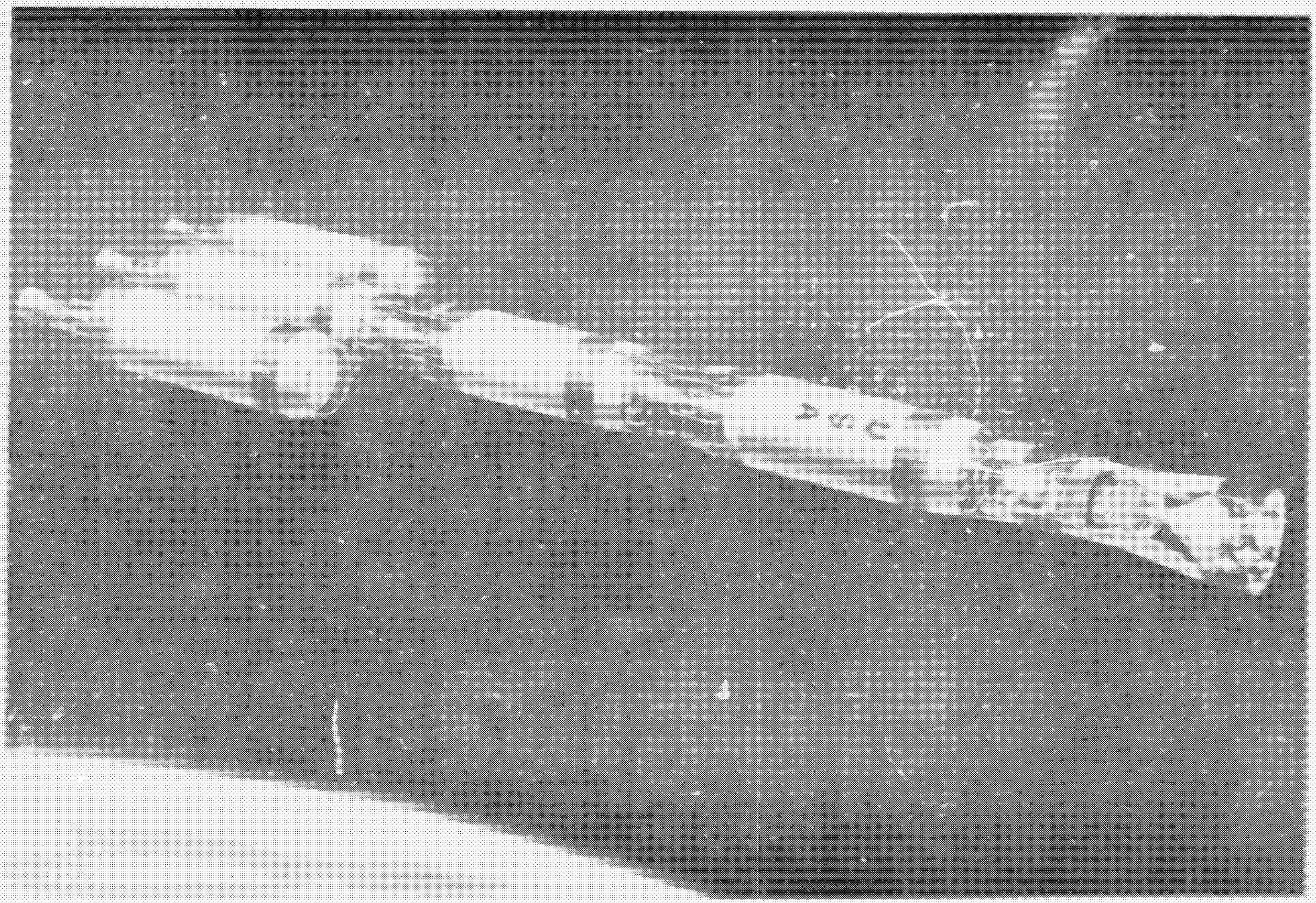


Figure 3.6-1. Manned Mars Space Vehicle

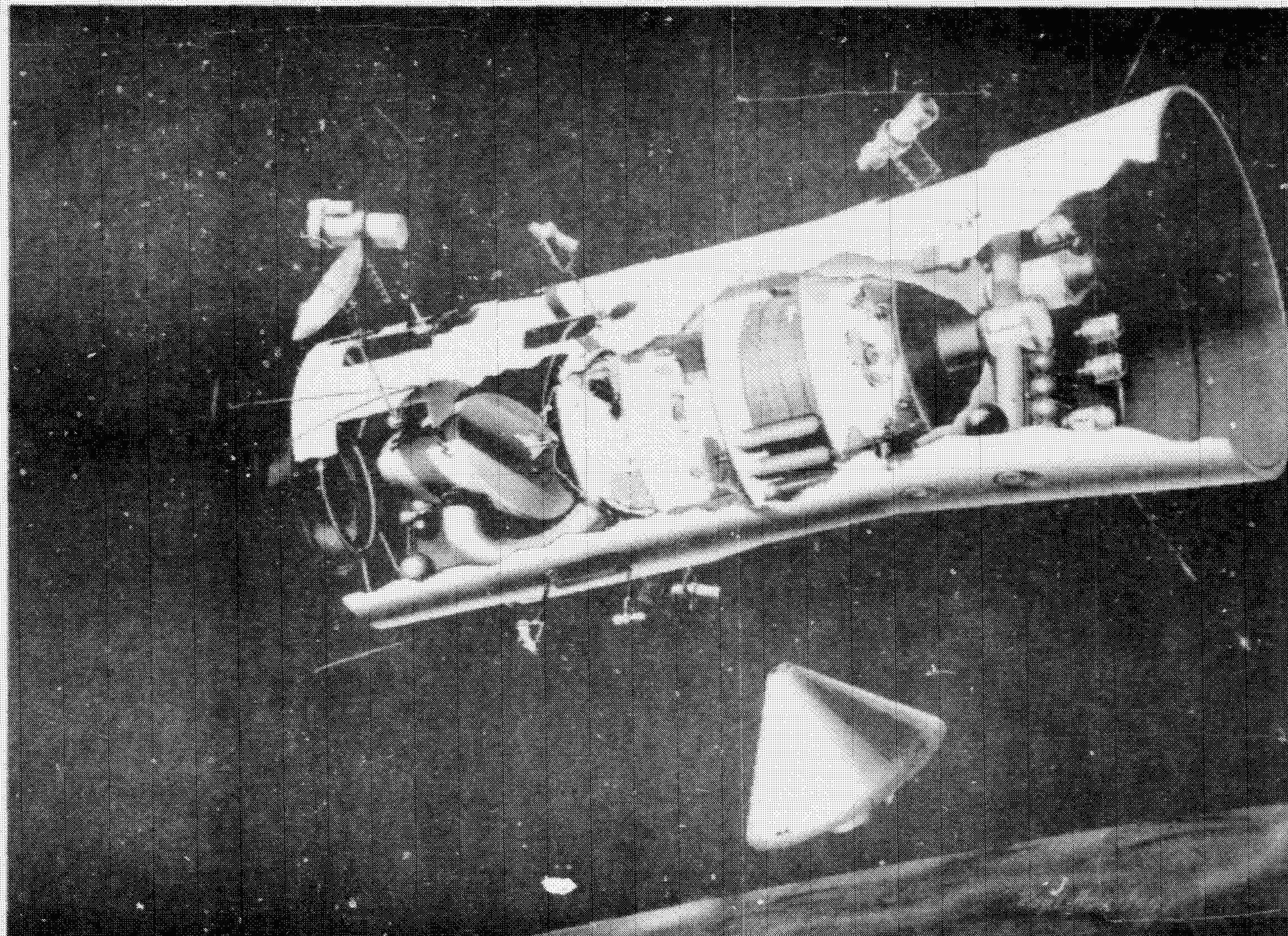


Figure 3.6-2. Mission Spacecraft

Table 3.6-1. Manned Planetary Mission Summary

MISSION	OBJECTIVES	MISSION ASSUMPTIONS & CONSTRAINTS
MANNED MARS LANDING	CONDUCT IN DEPTH SCIENCE PROGRAM DEALING WITH: • MARS PLANETOLOGY • EFFECTS OF MODIFYING FORCES • COMPOSITION • ENVIRONMENT • POSSIBLE LIFE FORMS	• EXAMINE CONJUNCTION AND OPPOSITION CLASS MISSIONS, 1994 LANDING • ELLIPTIC MARS PARKING ORBIT AT OPTION • SIX TO EIGHT MAN CREW • THREE TO FOUR MAN LANDER, ONE OR TWO PER MISSION • ONE ROVER PER LANDER, TWO-MAN CAPACITY' 100 kM TO RANGE • LONG LIFE INSTRUMENTATION SYSTEM OF "ALSEP" TYPE + METEOROLOGY STATION, PER LANDER • SCIENCE & SAMPLE RETURN CAPACITY OF 1000 kg

3.6.1.2 Mission Systems Description

3.6.1.2.1 Mission Options

Not applicable—manned Mars landing was the only mission considered.

3.6.1.2.2 Payload Descriptions

The spacecraft assemblv shown in figure 3.6-3 consists of three modules occupied by the crew during the course of the mission, connecting interstages, subsystems to provide operational capability, experiment equipment and sensors, and unmanned probes. Overall length is approximately 33m (108 ft) with maximum diameter of 10m (33 ft).

The forward interstage compartment is an unpressurized area that supports and encloses the Earth entry module, mission module subsystems, external experiment sensors, and the inbound midcourse propulsion system. A tunnel connects the EEM and mission module crew compartment to provide for pressurized transfer.

The mission module crew compartment provides the crew with a shirt-sleeve environment, quarters for living functions, space-vehicle operation capability, experiment laboratories, radiation shelter, and many of the subsystems required to support the above functions. The various functional areas and equipment are distributed on four decks. This compartment is occupied by six crewmen for the entire mission, except during the time when three crewmen descend to the Martian surface and during the Earth entry phase of the mission.

The aft interstage compartment is an unpressurized area having the shape of a truncated cone. This compartment houses the remainder of the mission module subsystem, external experiment sensors, and a portion of the unmanned probes. An airlock extends from the crew compartment to provide

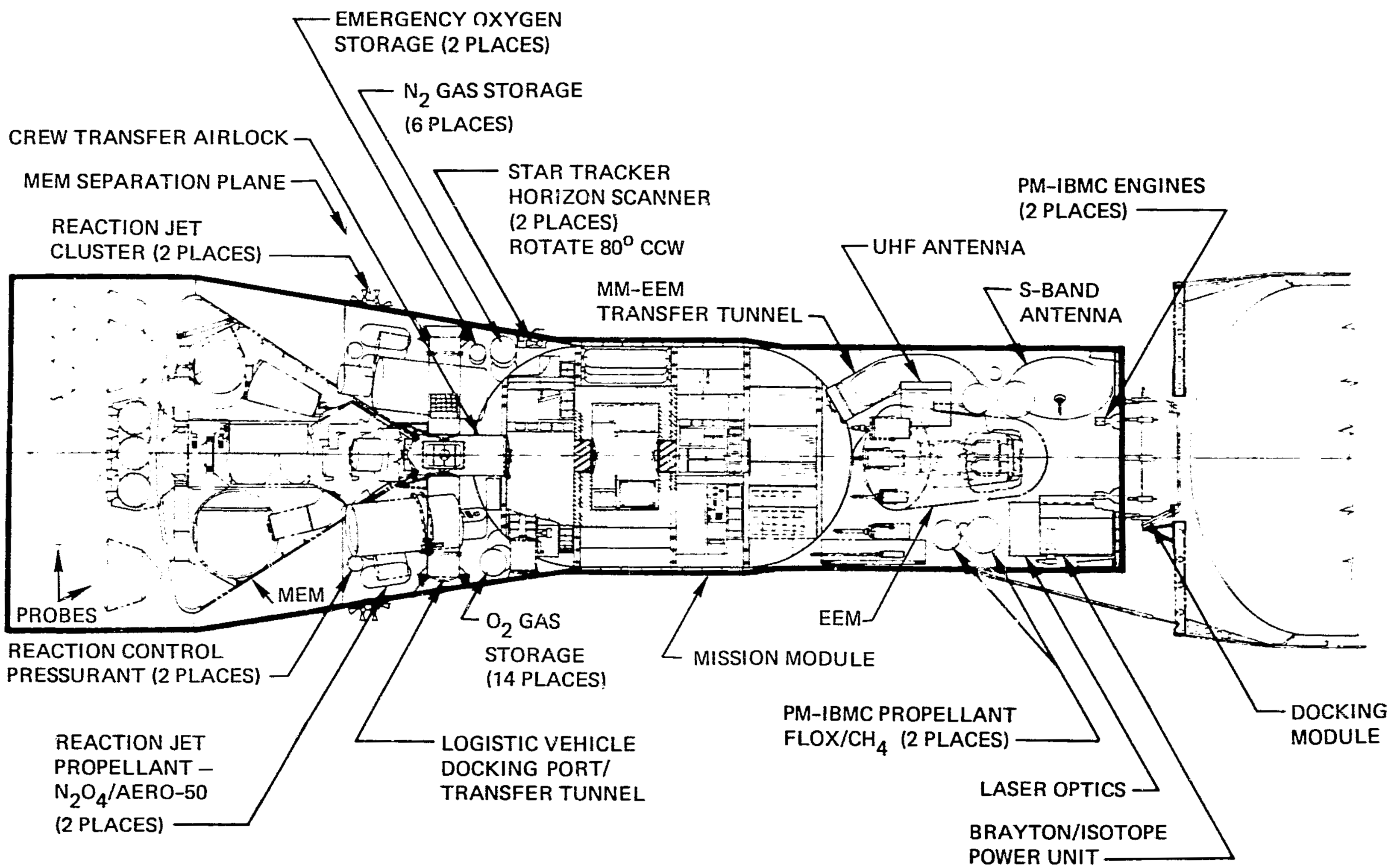


Figure 3.6-3. Mission Spacecraft

for pressurized trnasfer to the MEM and also provide EVA operation. Tunnels connect the airlock and logistics vehicle docking ports to provide additional pressurized transfer capability.

Located within the aft-most portion of the truncated-cone interstage is the Mars excursion module. The purpose of the MEM is to transfer three crewmen to the Martian surface, to provide living and operations quarters while on the surface, and to return the crewmen to the space vehicle. Unmanned probes occupy the 10m (33-foot) diameter aft cylindrical portion of the spacecraft.

3.6.1.2.2.1 Mission Modules

The primary functions of the mission module, shown in figure 3.6-3 are to provide shirt-sleeve-environment subsystems, experiment and vehicle operations control subsystems, the necessary structure to enclose and support the above, and to provide attachment to the MEM interstage and the Mars orbit-departure propulsion module (PM-3). The recommended mission module has sufficient volume for equipment and expendables suitable for Mars, Venus, and Mars/Venus swingby missions of durations up to approximately 1,100 days. In addition, the mission module serves as the living and operations center for the assembly, test, and checkout crew while the interplanetary space vehicle is being assembled in Earth orbit. The mass of the mission module is 37 100 kg (69,900 lb) for a 450-day mission and 47 000 kg (103,600 lb) for a 1000-day mission.

3.6.1.2.2.2 Mars Excursion Module

The Mars excursion module (MEM) is a 9.1m- (30-ft) diameter, Apollo-shape, three-man vehicle (fig. 3.6-4). It brings three men to the Martian surface, houses them for 30 days, and then returns them to the orbiting spacecraft. External solid rockets provide the impulse for deorbit. FLOX/CH_4 propellants are used for ascent and the final stages of descent in the configuration described. Other propellants could be used. The ascent delta V to a circular orbital altitude of 1 000 km (540 nmi) is 5 300 m/sec (17,300 fps). The outer shell and laboratory are staged at the Mars surface. Only the central core ascends to the space vehicle with the crew. Upon transfer of crew and Mars surface samples to the spacecraft, the MEM ascent stage is jettisoned and remains in Mars orbit. The MEM mass is 33 100 kg (73,000 lb).

3.6.1.2.2.3 Biconic EEM Configuration

The Earth entry module (EEM) configuration is a six-man blunted biconic (fig. 3.6-5). In order to maintain a high degree of commonality, the EEM design is fixed for all missions. The heat shield is designed by the highest Earth entry velocity, approximately 18 000 m/sec (60,000 fps) as defined by the opposition mission. The EEM mass is 6 850 kg (15,100 lb) for a 450-day mission and 5 260 kg (11,600 lb) for a 1000-day mission.

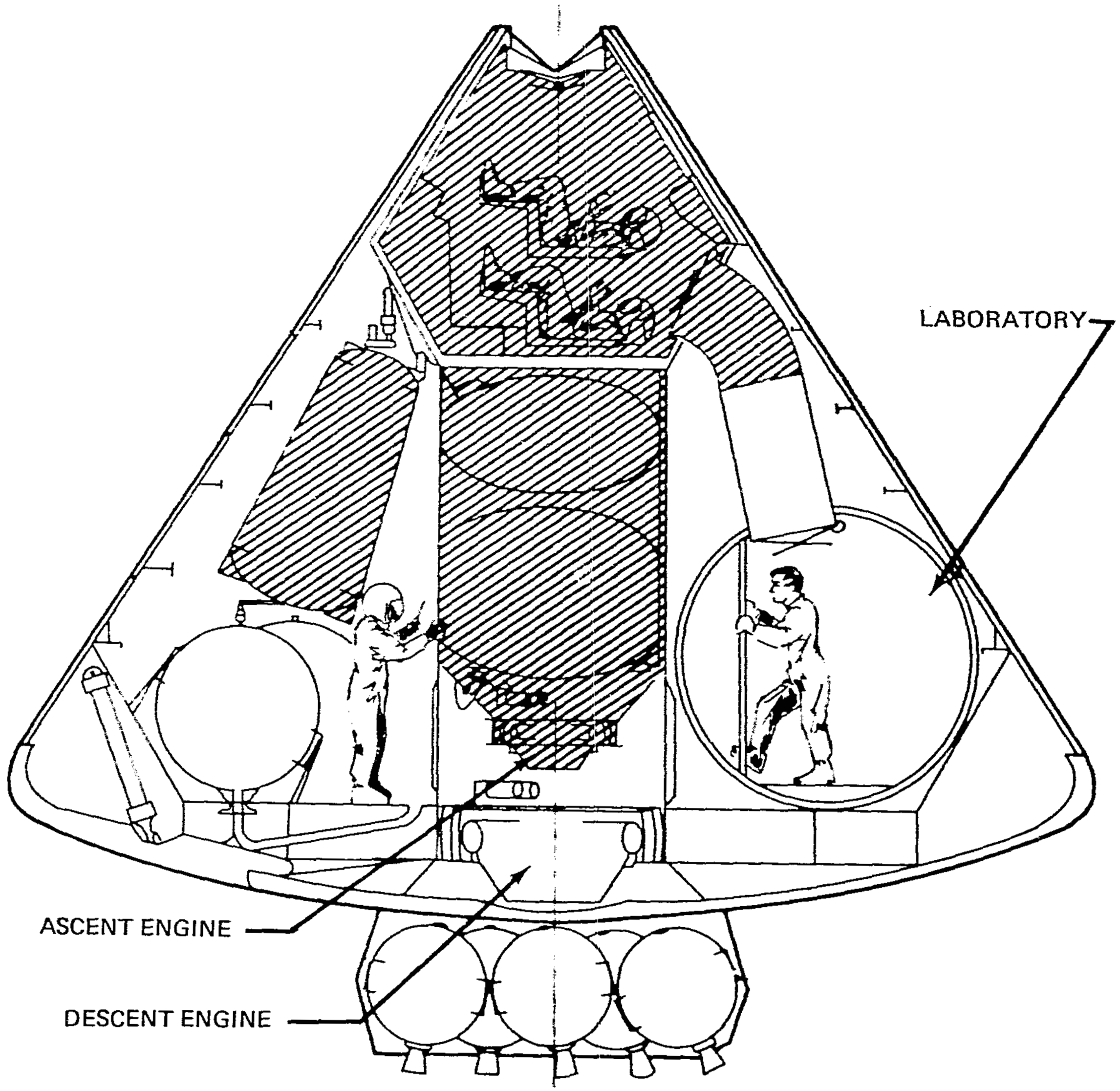


Figure 3.6-4. Mem Configuration Three-Men–30 Days–540 N MI Orbit

The EEM systems are designed for 1 day occupancy prior to Earth entry. Final descent is made by parachutes with a capability of landing either on land or water.

3.6.1.2.2.4 Probes and Interstages

The mass of the probes and subsatellites carried is 8 890 kg (19,600 lb). The total interstage mass is 9 100 kg (20,000 lb).

3.6.1.2.2.5 Crew Rotation and Resupply Payloads

Not applicable.

3.6.1.2.2.6 Mass Summary

Payload masses are summarized in table 3.6-2.

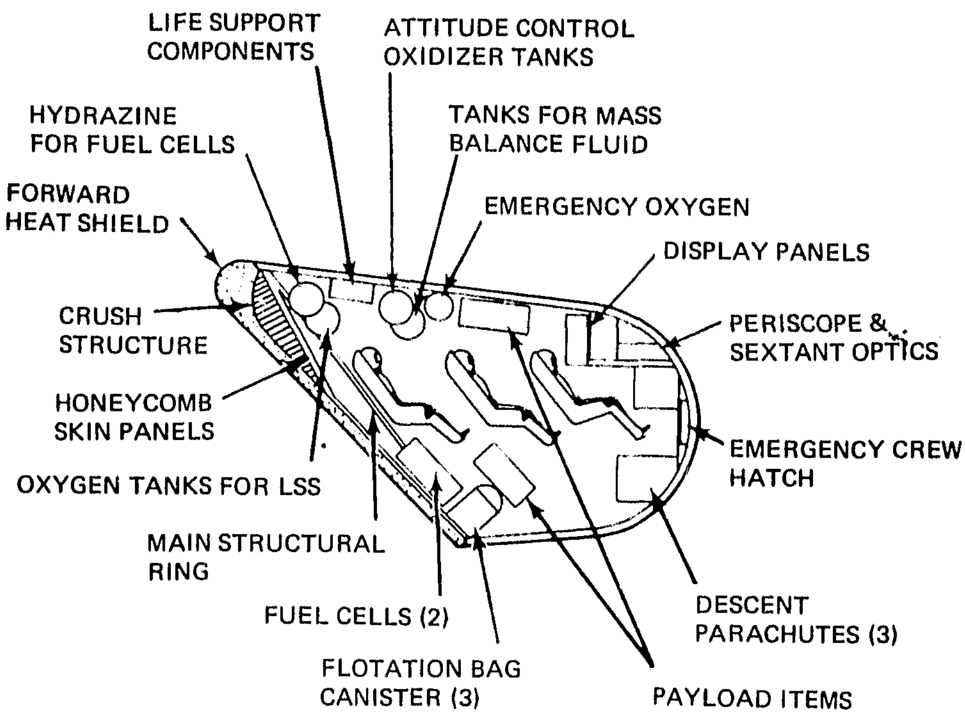


Figure 3.6-5. Biconic EEM Configuration Six-Man Crew

Table 3.6-2. Manned Mars Payload Masses

Item	Mars Opposition (450 Days)		Mars Conjunction (1000 Days)	
	Kg	Lb	Kg	Lb
MM (includes experiments)	31 700	69,900	47 000	103,600
MEM	33 100	73,000	33 100	73,000
EEM	6 850	15,100	5 260	11,600
Probes	8 890	19,600	8 890	19,600
Interstages	9 100	20,000	9 100	20,000
Total	89 640	197,600	103 350	227,800

3.6.1.2.2.7 Pickup Points and Transportation Constraints

The payload described is designed to be launched to Earth orbit as a single unit by a HLLV. In orbit it will be joined with the orbit transfer system.

No transportation constraints were found except mission profile constraints and options. These are discussed in para. 3.6.1.3.2.3.

3.6.1.2.3 Transfer and Storage

Not applicable.

3.6.1.2.4 Orbital Assembly, Maintenance, and Modification

The mission vehicles described are designed to be serviced and maintained by the mission crew.

3.6.1.3 Transportation Requirements

3.6.1.3.1 Payload Delivery Points

Payload delivery points are as follows:

Space Vehicle Assembly	Low Earth Orbit
Mission Orbit	Mars Orbit
Planet Exploration	Mars Surface
Crew & Science Return	Mars orbit to Earth orbit or direct entry and landing

3.6.1.3.2 Payload Delivery Options

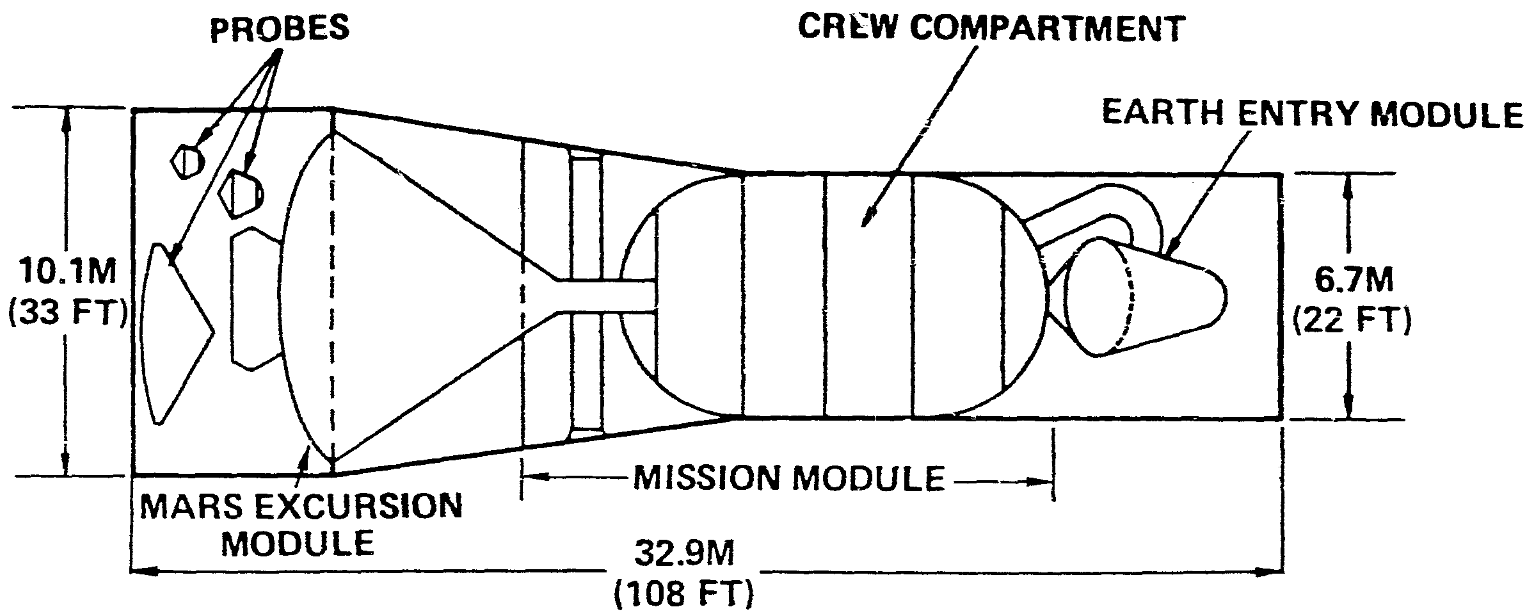
3.6.1.3.2.1 Mission Vehicle

Mass and size characteristics of the payload, including applicable mass growth allowances, are summarized in figure 3.6-6. Payload masses for each principal transportation operations are summarized in figure 3.6-7.

3.6.1.3.2.2 Operational Constraints

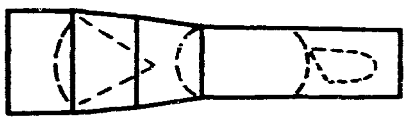
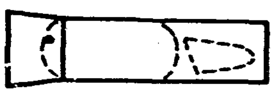
Constraints result from mission profile characteristics. Mars stopover mission opportunities occur in two principal varieties (1) conjunction long duration, long stay at Mars, low energy, and (2) opposition: short duration, short stay (optimum = 0) at Mars, high energy. Intermediate duration Venus swingby opportunities also occur at less frequent intervals.

The missions characteristics described span the range of energy requirements of the various Mars opportunities existing during the typical Mars synodic cycle from 1975 to 1990 (table 3.6-3). The



PAYLOADS	MASS			
	MARS OPPOSITION (450 DAYS)		MARS CONJUNCTION (1,000 DAYS)	
	KG	LB	KG	LB
MISSION MODULE (INCL EXPERIMENTS)	37 600	82,900	52 900	116,600
MARS EXCURSION MODULE	43 200	95,300	43 200	95,300
EARTH-ENTRY MODULE	7 900	17,400	6 300	13,900
PROBES	11 100	24,500	11 100	24,500
INTERSTAGES	9,500	21,000	9 500	21,000
TOTAL	109 300	241,100	123 000	271,300

Figure 3.6-6. Manned Mars Spacecraft System Elements

MISSION PHASE	PAYLOAD MASS*			
	MARS OPPOSITION		MARS CONJUNCTION	
	KG	LB	KG	LB
1. TRANS MARS INJECTION/ MARS ORBIT INJECTION 	109 591	241,100	123 318	271,300
2. TRANS EARTH INJECTION 	50 455	111,000	64 227	141,300
3. EARTH ORBIT INJECTION 	7 909	17,400	6 300	13,900

* EXCLUDES ANY AEROBRAKING PROVISIONS

Figure 3.6-7. Manned Mars Transportation Requirements

planetary stay time is 40 days for opposition missions and selected at minimum energy for the conjunction mission (approximately 500 days at Mars). The requirements shown on the chart were derived by developing trajectories that provided the minimum initial mass in Earth orbit. Those requirements with the greatest impact on the spacecraft were the Earth-entry velocity and the mission time.

Elliptic orbits at Mars may reduce energy requirements below those stated. For this to be true, an elliptic orbit must be found that allows near periapsis-to-periapsis transfer without excessive plane change.

Table 3.6-3. Mission Parameters Variations

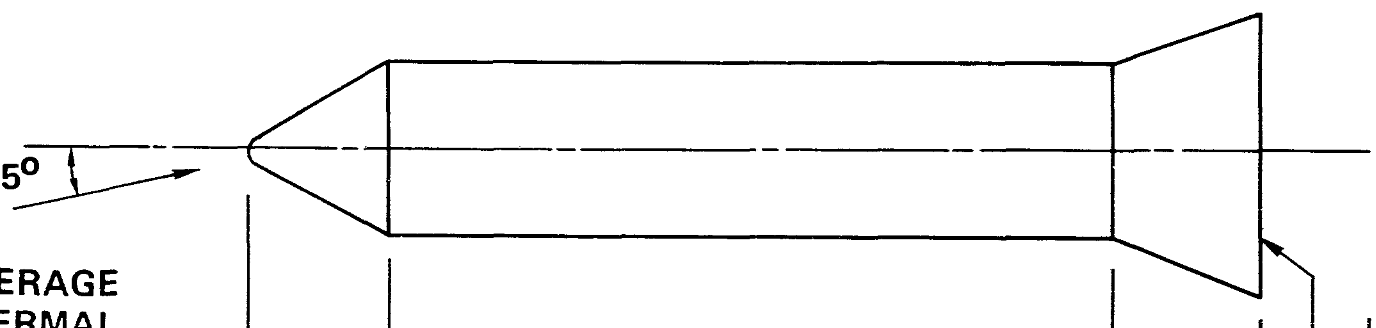
EARTH DEPARTURE ΔV	3600-5000 M/SEC	(11,700-16,700 FPS)
MARS ARRIVAL ΔV (1000 KM CIRCULAR ORBIT)	2100-5300 M/SEC	(6,950-17,400 FPS)
MARS DEPARTURE ΔV (1000 KM CIRCULAR ORBIT)	1900-5800 M/SEC	(6,320-19,000 FPS)
EARTH ENTRY VELOCITY	11,500-18,300 M/SEC	(38,000-60,200 FPS)
TOTAL MISSION TIME	(OPPOSITION) 460 – 1,040 DAYS	(CONJUNCTION)
PLANET STAYTIME	(OPPOSITION) 40 – 580 DAYS	(CONJUNCTION)

3.6.1.4 Mission/Transportation Modes and Operations

3.6.1.4.1 Transportation Options

The principal transportation candidates considered for the three major mission velocity requirements are as follows:

- An all nuclear system (NNN) consisting of three separate stages.
- An all chemical system (CCC) using three separate LO_2/LH_2 stages.
- A chemical/aerobraking/chemical system (CAC) using a LO_2/LH_2 stage for TMI, aerobraking for MOI and a LO_2/LH_2 stage for TEI.
- The aerobraking portion of this vehicle is a cone-cylinder-flare design as shown in figure 3.6-8. Enclosed within the aerobraking configuration is the mission spacecraft and TEI stage. A fixed deflection in the flare forces the vehicle to fly at an angle of attack consistent with the desired lift-to-drag ratio. The direction of the lift vector is controlled by rolling the vehicle with



AVERAGE THERMAL PROTECTION	NOSE	CYLINDER	FLARE	BASE
KG/M^2	48.4	17.0	30.0	9.7
(LB/FT^2)	(10.0)	(3.5)	(6.2)	(2.0)

LIFT/DRAG = 1.0

M/C_DA = 3930 KG/M^2 (805 LB/FT^2)

V_E = 6860 MPS (22,500 FPS)

MARS VM – 8 ATMOSPHERE

CIRCULARIZATION ΔV = 280 MPS (920 FPS)

Figure 3.6-8. Aerobraking Configuration Characteristics

reaction jets. The nose, flare, and lower half of the cylinder are protected by ablation material. The upper half of the cylinder and the base of the flare are protected by Rene' 41 radiation material. The average unit weights are determined for the conditions shown.

The only Earth launch vehicle considered is a HLLV using modified components of the shuttle. The space shuttle would transport crews and supplies during the space vehicle assembly phase.

3.6.1.4.2 Representative Transportation Mode and System

3.6.1.4.2.1 Transportation Sequence

Typical sequences and operations associated with an all nuclear transportation system are illustrated in figure 3.6-9. The principal phases during the mission include Earth orbit operations, Earth-Mars transit, Mars operations and Mars-Earth transit.

During the Earth orbital operation phase the mission spacecraft will be delivered to orbit with a HLLV. Delivery of seven nuclear propulsion modules and two drop tanks and two fueling tankers will complete the assembly and fueling of the manned Mars space vehicle. During the last few weeks of this phase, the mission crew will be delivered to complete the checkout of the vehicle.

The Earth-Mars transit phase consists of Earth orbit departure, mid course corrections and injection into a high Mars orbit using the second nuclear stage system. This nuclear system is jettisoned and the vehicle transferred down to a lower operational orbit.

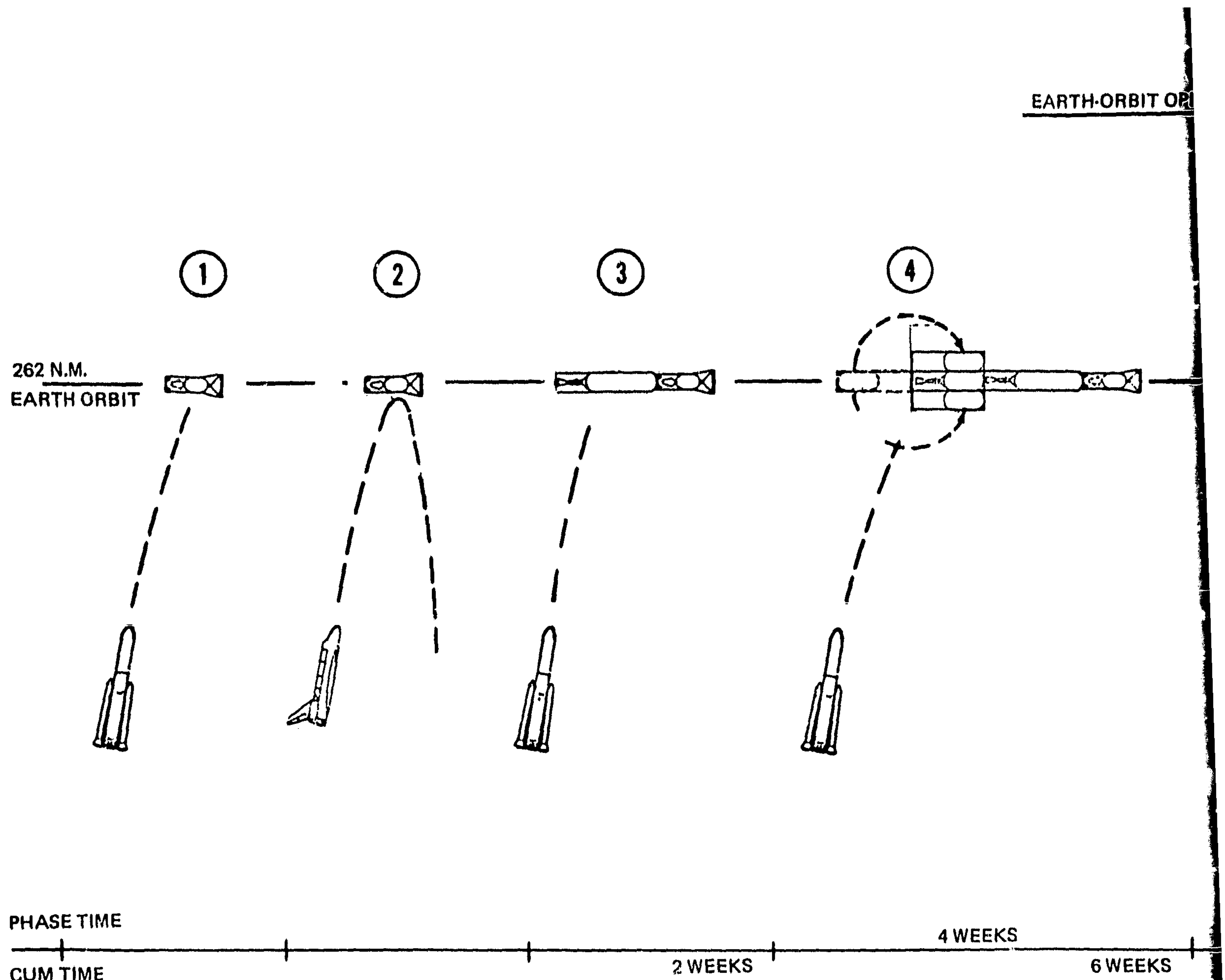
Mars orbit operations include launch of unmanned probes to assist in selecting the MEM landing site, the 30 day excursion of the MEM to the surface and launching of unmanned scientific probes.

Mars-Earth transit consists of departure from Mars orbit using the last nuclear stage, inbound midcourse corrections and separation of the Earth entry module including the crew approximately one day before scheduled entry. Should a high speed re-entry not be desired, a storable propulsion system could be used to place a small manned vehicle back into Earth orbit where retrieval could be accomplished with the Shuttle.

Use of an all chemical transportation system would result in generally the same operations as for the all nuclear system.

3.6.1.4.2.2 Transportation

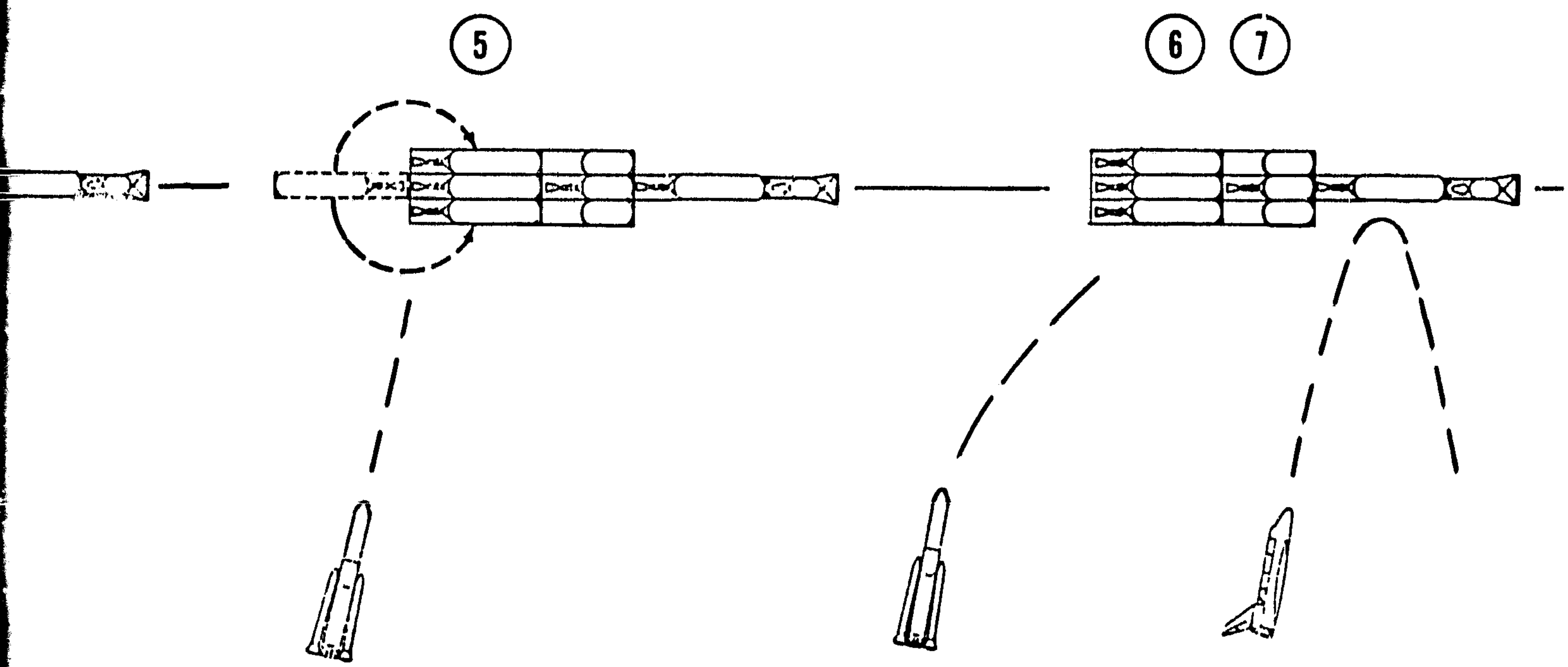
Parametrics were not developed. Sizing for the options is compared in paragraph 3.6.1.4.3.2



1	2	3	4
• DELIVER UNMANNED SPACECRAFT TO ORBIT WITH HLV	• DELIVER ASSEMBLY AND CHECKOUT CREW WITH SS	• DELIVER TEI STAGE TO ORBIT WITH HLV • DOCK AND ASSEMBLE TEI STAGE TO SPACECRAFT • CHECK OUT SUB-ASSEMBLY	• DELIVER MOI STAGE TO ORBIT WITH HLV, (2 LAUNCHES)

FOLDOUT FRAME 1

EARTH-ORBIT OPERATIONS



4 WEEKS | 3 WEEKS | 2 WEEKS

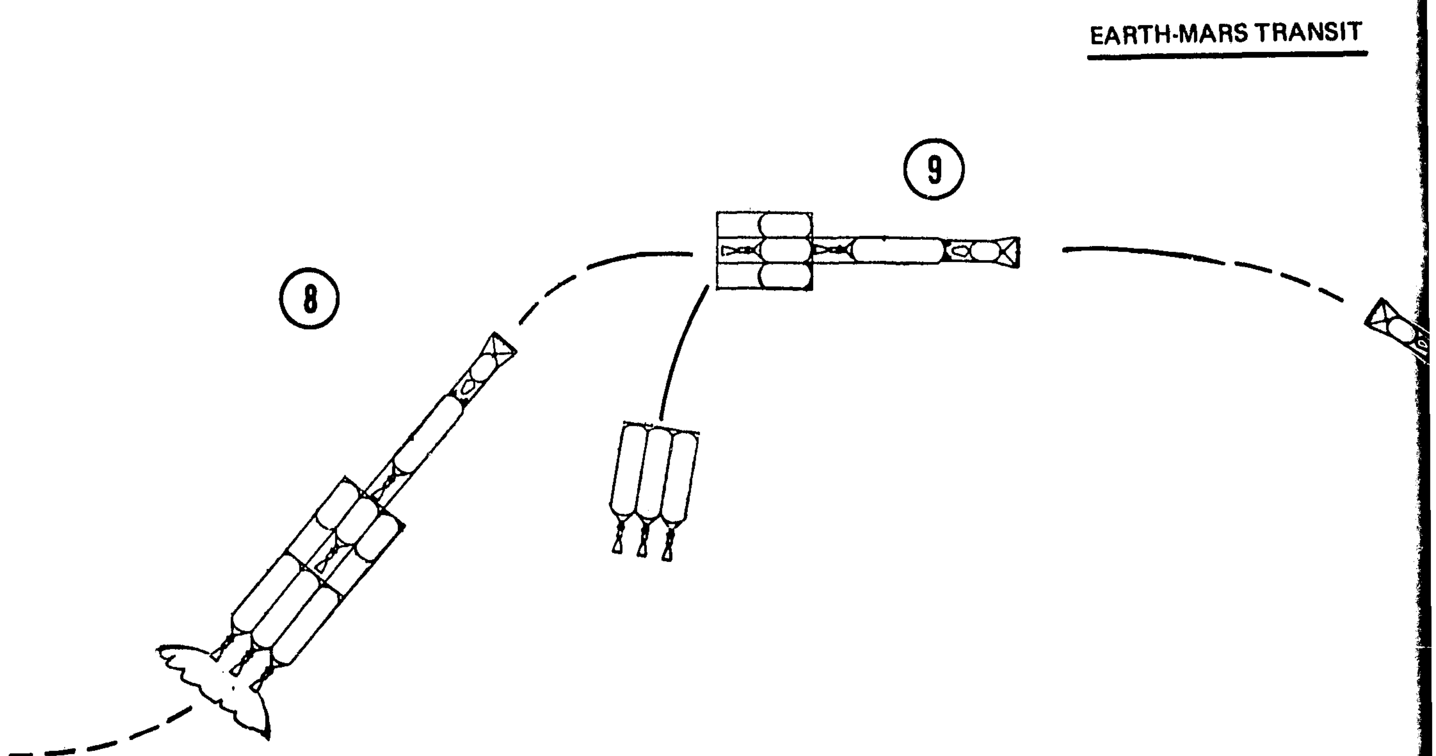
6 WEEKS | 10 WEEKS | 13 WEEKS | 15 WEEKS

5 | 6 7

E TO ORBIT CHES)	• DELIVER TMI STAGE TO ORBIT WITH HLV, (5 LAUNCHES)	• DELIVER TANKER WITH HLV, (3 LAUNCHES) • FUEL OTV • DISPOSE OF EMPTY TANKER TANKS	• DELIVER MISSION CREW TO SPACECRAFT WITH SS • TOP OFF PROPELLANT • RETURN ASSEMBLY AND CHECKOUT CREW TO EARTH SURFACE

Figure 3.6-9 Manned Mars Landing Mission Nuclear-Nuclear-Nuclear (Sheet 1)

FOLDOUT FRAME 2

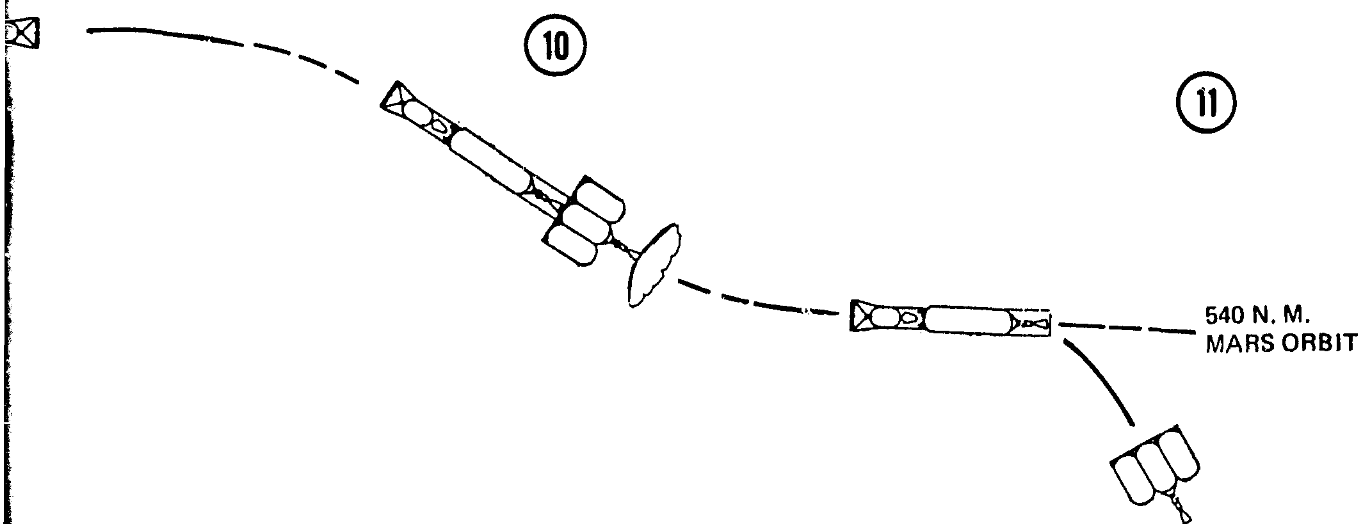


PHASE TIME	< 3 HOURS	190-290 DAYS	
CUM TIME			

	8	9	
	• JETTISON TMI STAGE INTERSTAGE AND SHIELDING • TMI BURN	• JETTISON TMI STAGE • COAST • THREE MIDCOURSE CORRECTIONS • CONDUCT SCIENTIFIC/ENGINEERING EXPERIMENTS	• JETTISON M SHIELDING • INJECT INTO

FOLDOUT FRAME /

EARTH-MARS TRANSIT



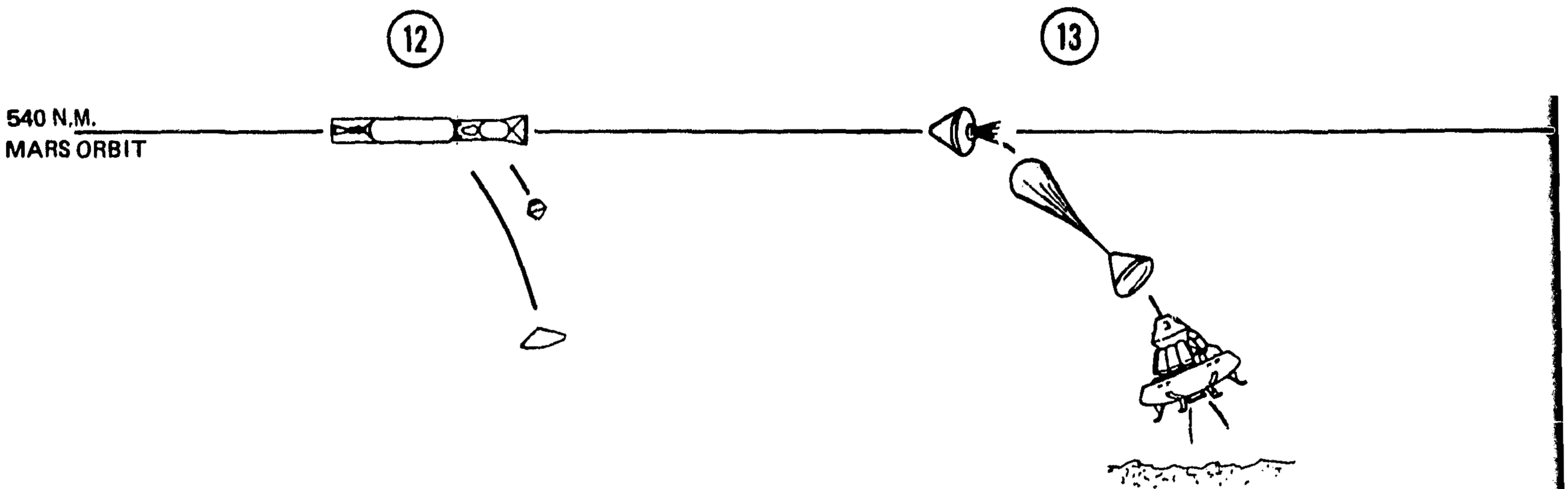
S

	10	11
)RRECTIONS ENGINEERING	• JETTISON MOI STAGE INTERSTAGE AND SHIELDING • INJECT INTO 640 N.M. MARS PARKING ORBIT	• JETTISON MOI STAGE • TRANSFER TO MARS 540 N.M. CIRCULAR OPERATIONAL ORBIT USING ORBIT TRIM PROPULSION

Figure 3.6-9 . Manned Mars Landing Mission Nuclear/Nuclear/Nuclear (Sheet 2)

FOLDOUT FRAME 2

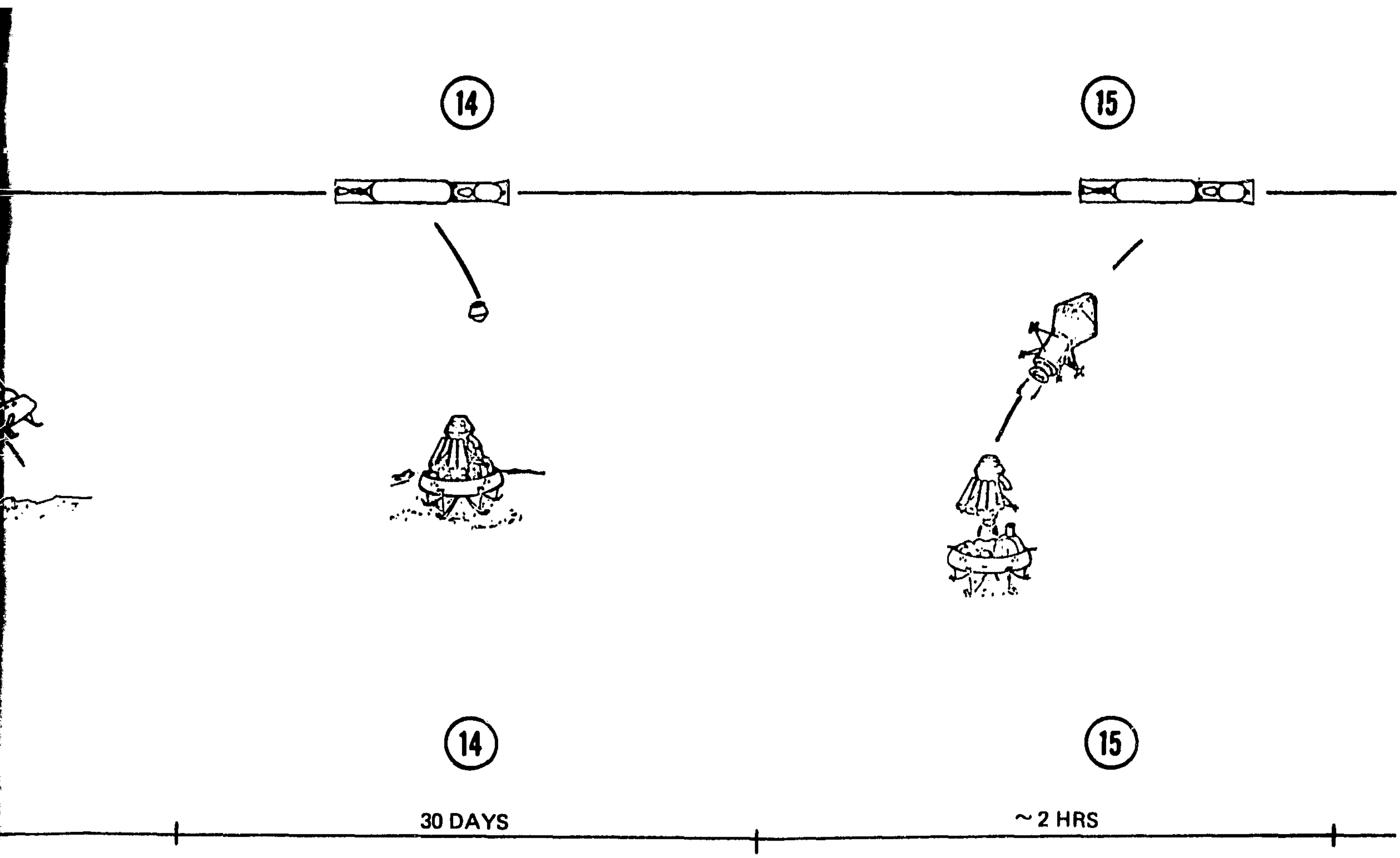
MARS OPERATIONS



	12	13
PHASE TIME CUM TIME	2 TO 5 DAYS	~ 2 HRS
	• LAUNCH ENGINEERING PROBES • SURVEY FOR MEM LANDING SITES	• DEORBIT MEM • AEROBRAKE USING BALLUTE AND HEAT SHIELD • LAND, USING DESCENT ENGINE

FOLDOUT FRAME

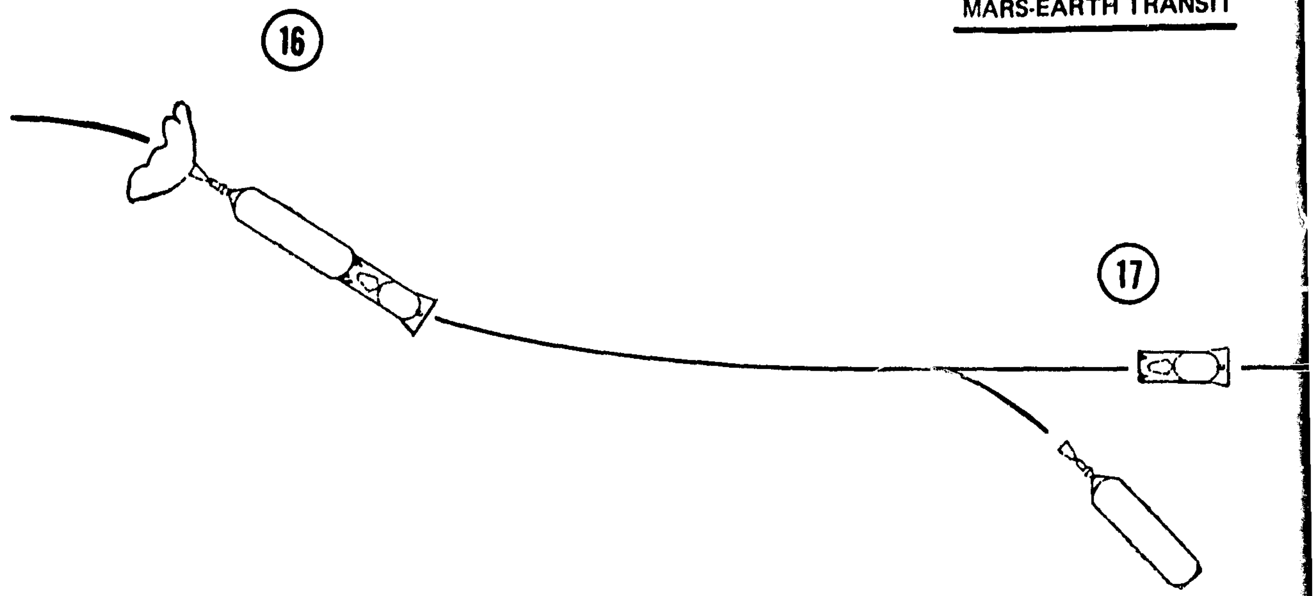
MARS OPERATIONS



• CONDUCT ORBITAL SCIENCE • LAUNCH SCIENTIFIC PROBES • COLLECT SURFACE DATA AND SAMPLES FOR 30 DAYS	• MEM TWO-STAGE ASCENT • RENDEZVOUS AND DOCK WITH SPACECRAFT

Figure 3.6-9 Manned Mars Landing Mission-Nuclear/Nuclear/Nuclear (Sheet 3)

MARS-EARTH TRANSIT

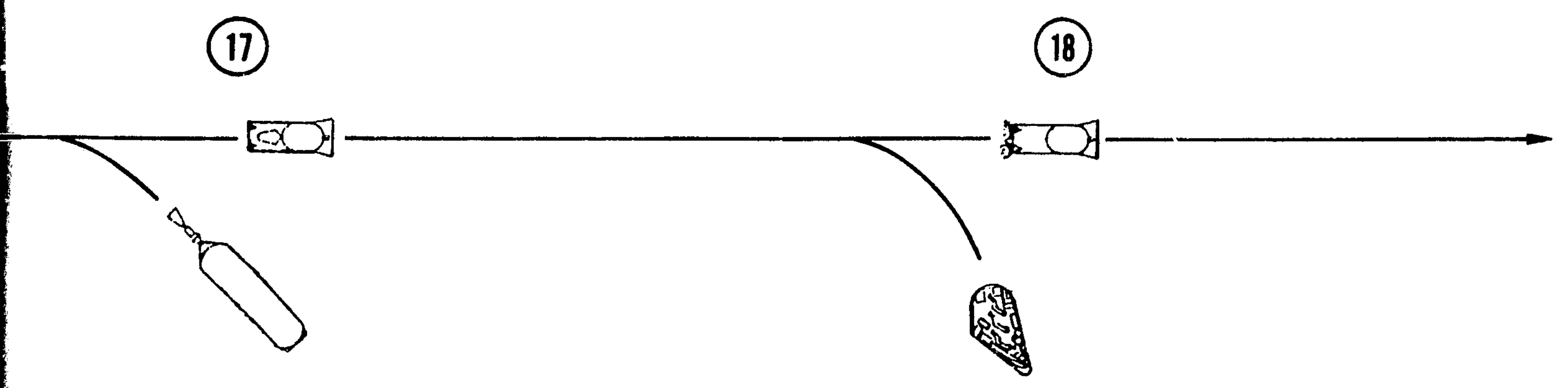


PHASE TIME — 210-250 DAYS
CUM TIME

16	17
• JETTISON TEI STAGE INTERSTAGE AND SHIELDING • TEI BURN	• JETTISON TEI STAGE • COAST • THREE MIDCOURSE CORRECTIONS • ANALYZE MARS SAMPLES AND EXPERIMENT DATA

FOLDOUT FRAME

MARS-EARTH TRANSIT



210-250 DAYS	< 1 DAY
	460-550 DAYS

17	18
• JETTISON TEI STAGE • COAST • THREE MIDCOURSE CORRECTIONS • ANALYZE MARS SAMPLES AND EXPERIMENT DATA	• SEPARATE EARTH ENTRY MODULE FROM MISSION MODULE • DIRECT REENTRY RETURN AND WATER RECOVERY OF EEM • DIVERT MISSION MODULE TO NON-EARTH-IMPACT TRAJECTORY

Figure 3.6-9 Manned Mars Landing Mission Nuclear/Nuclear/Nuclear (Sheet 4)

 (REVERSE IS BLANK)

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3.6.1.4.2.3 Operational Factors

Mission Profiles--Profile data were summarized in paragraph 3.6.1.3.2.3.

Crew Involvement and Timelines--Crew involvement will simplify the Mars orbit rendezvous required at the completion of the surface missions. Crew timelines do not have any identified effect on transportation requirements.

Control Functions and Requirements--Precision navigation and targeting are required for trans-Mars injection; Mars orbit insertion; Mars landing, ascent, and rendezvous; trans-Earth injection; and Earth entry and landing.

Network Support--The manned Mars landing mission will demand at least daily network support during the transplant coast phases, primarily to monitor systems status and alert the crew to any needed corrective action. Continuous support, except for communications blackouts due to occultation by Mars, will be needed during the Mars stay. Communications procedures must deal with the 20 to 30 minute signal round trip time delay from Earth to Mars and back.

3.6.1.4.2.4 Earth Launch Requirements Summary

Earth launch requirements are compared in paragraph 3.6.1.4.3.3.

3.6.1.4.3 Transportation Options Comparison and Evaluation

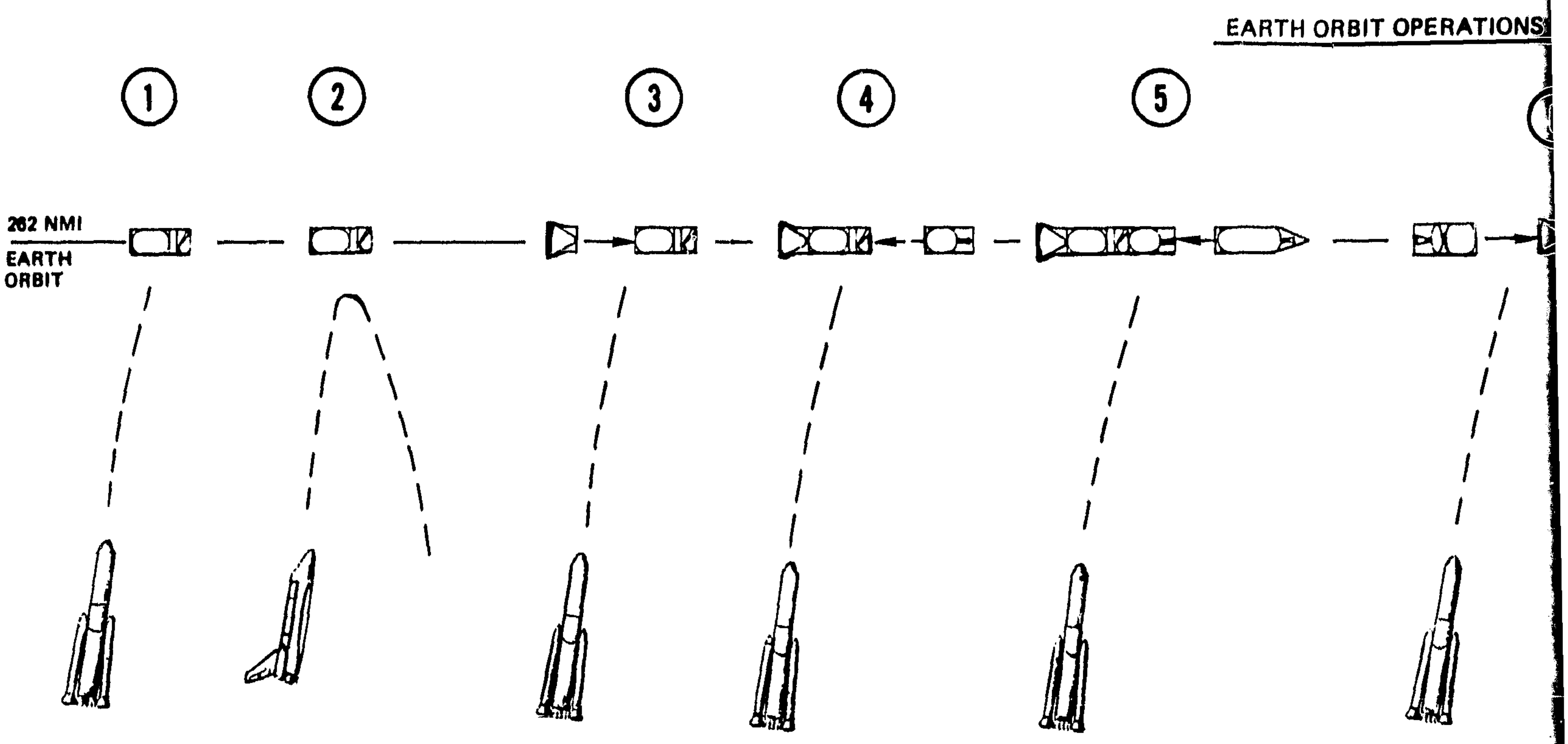
3.6.1.4.3.1 Alternate Sequence Description

The CAC mission sequence is illustrated in figure 3.6-10. The major difference in this sequence is the use of aerobraking for the MOI maneuver. The most significant propulsive maneuver during the MOI phase would be the transfer from the entry altitude up to the operational altitude of 1 000 km (540 n. mi.).

3.6.1.4.3.2 Size and Performance Comparison

Performance capability of the candidate transportation systems has been obtained from Boeing IMISCD study. The comparison of the initial mass in Earth orbit (IMIEO) for three mission classes is shown in figure 3.6-11. The opposition mission has the shortest duration but the highest energy requirements. Conjunction and Venus swingby have fairly comparable energy requirements for many mission opportunities but the swingby mission has a duration of 600 days versus 1,000 days.

In all missions, the all nuclear system results in a significant reduction in IMIEO compared to the all chemical system. However, should a nuclear program not be undertaken, the aerobraking concept provides a reasonable IMIEO for the opposition mission with chemical propulsion.



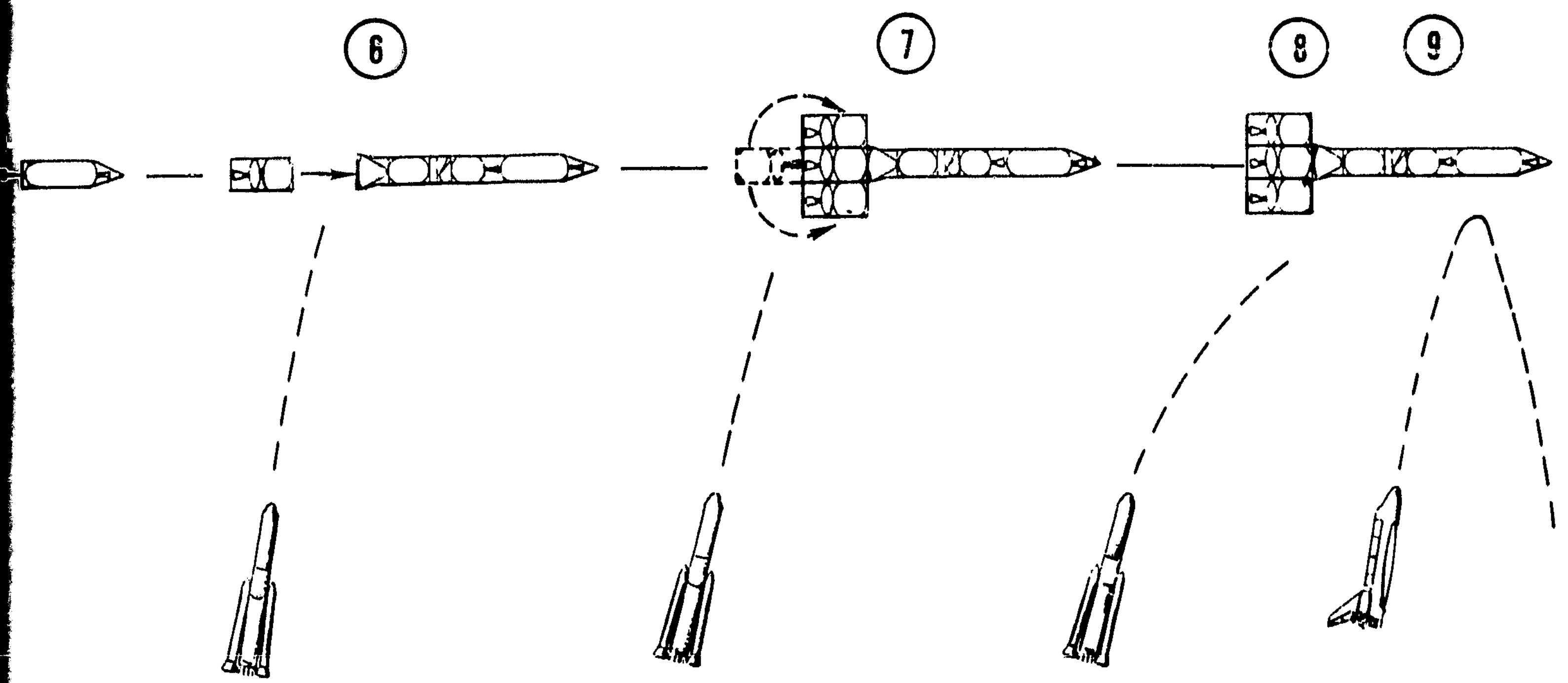
PHASE TIME

CUM TIME: 0 | 2 WEEKS | 5 WEEKS | 6 WEEKS

1	2	3	4	5
• DELIVER UNMANNED MISSION MODULE (MM) AND EARTH ENTRY MODULE (EEM) TO ORBIT WITH HLV	• DELIVER ASSEMBLY AND CHECKOUT CREW WITH SS	• DELIVER MARS EXCURSION MODULE (MEM) WITH HLV • DOCK AND ASSEMBLE MEM TO MM AND EEM TO FORM SPACECRAFT	• DELIVER TEI STAGE No. 2 TO ORBIT WITH HLV • DOCK AND ASSEMBLE TEI STAGE 2 TO SPACECRAFT	• DELIVER TEI STG 1 AND NOSECONE TO ORBIT WITH HLV • DOCK AND ASSEMBLE TO TEI STG 2 TO FORM AEROBRAKING VEHICLE

FOLDOUT FRAME

EARTH ORBIT OPERATIONS

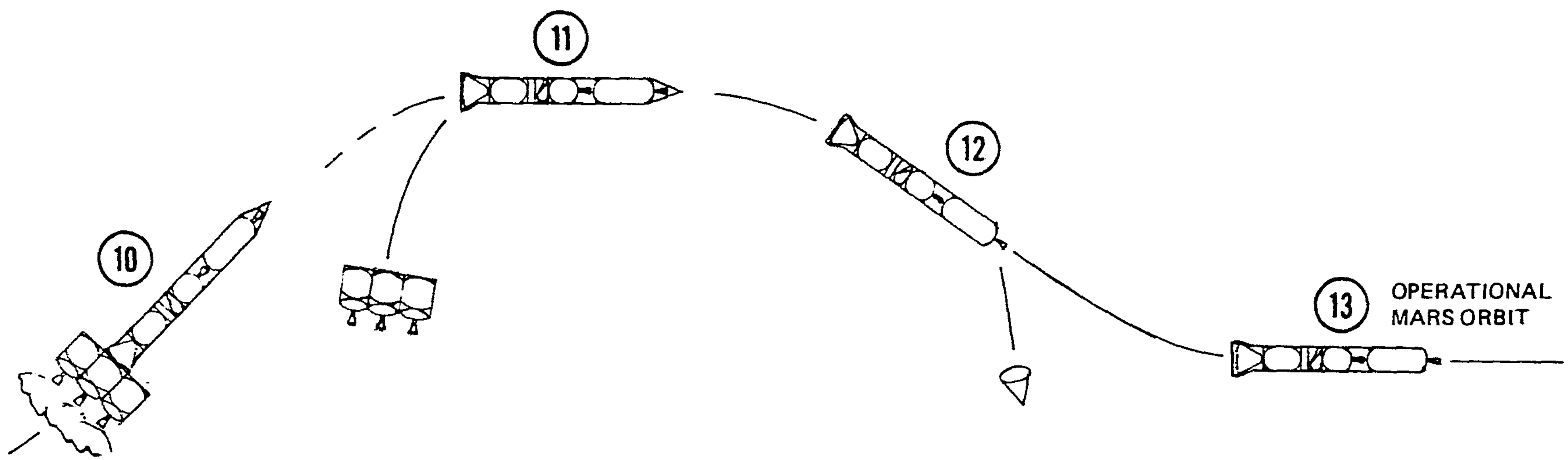


6 WEEKS	9 WEEKS	17 WEEKS	22 WEEKS	23 WEEKS
5	6	7	8	9
• DELIVER TEI STG 1 AND NOSECONE TO ORBIT WITH HLV • DOCK AND ASSEMBLE TO TEI STG 2 TO FORM AEROBRAKING VEHICLE	• DELIVER TMI CORE MODULE TO ORBIT WITH HLV • DOCK AND ASSEMBLE MODULE TO AEROBRAKING VEHICLE	• DELIVER FINAL FOUR TMI MODULES WITH HLV	• DELIVER TANKER WITH HLV (5 LAUNCHES) • FUEL TMI AND TEI STAGES • DISPOSE OF EMPTY TANKERS	• DELIVER MISSION CREW TO SPACECRAFT WITH SS • TOP OFF PROPELLANTS • RETURN ASSEMBLY AND CHECKOUT CREW TO EARTH SURFACE

Figure 3.6-10 Manned Mars Landing Mission-Chemical/Aerobraking/Chemical (Sheet 1)

FOLDOUT FRAME
2

EARTH-MARS TRANSIT OPERATIONS



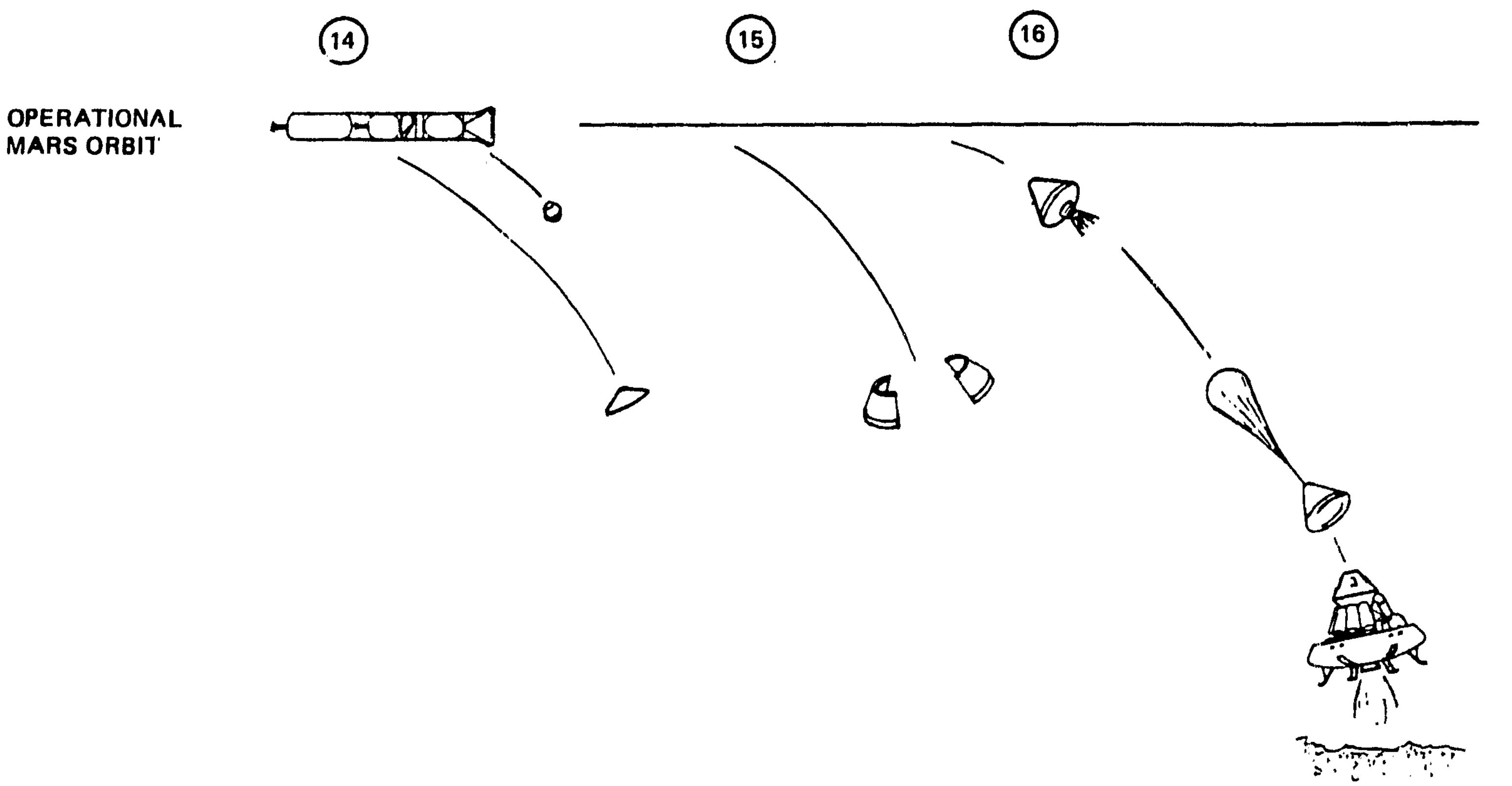
PHASE TIME — 190-290 DAYS

CUM TIME

10	11	12	13
• JETTISON TMI STAGE INTERSTAGE AND METEOROID SHIELDING • TMI BURN	• JETTISON TMI STAGE • COAST • THREE MIDCOURSE CORRECTIONS • CONDUCT SCIENTIFIC/ENGINEERING EXPERIMENTS	• AEROBRAKE INTO MARS ATMOSPHERE • JETTISON NOSE CONE • CIRCULARIZE INTO HIGH MARS PARKING ORBIT USING ORBIT TRIM PROPULSION	• TRANSFER TO 540 NMI CIRCULAR OPERATIONAL ORBIT USING ORBIT TRIM PROPULSION

Figure 3.6-10 Manned Mars Landing Mission-Chemical/Aerobraking/Chemical (Sheet 2)

MARS OPERATIONS



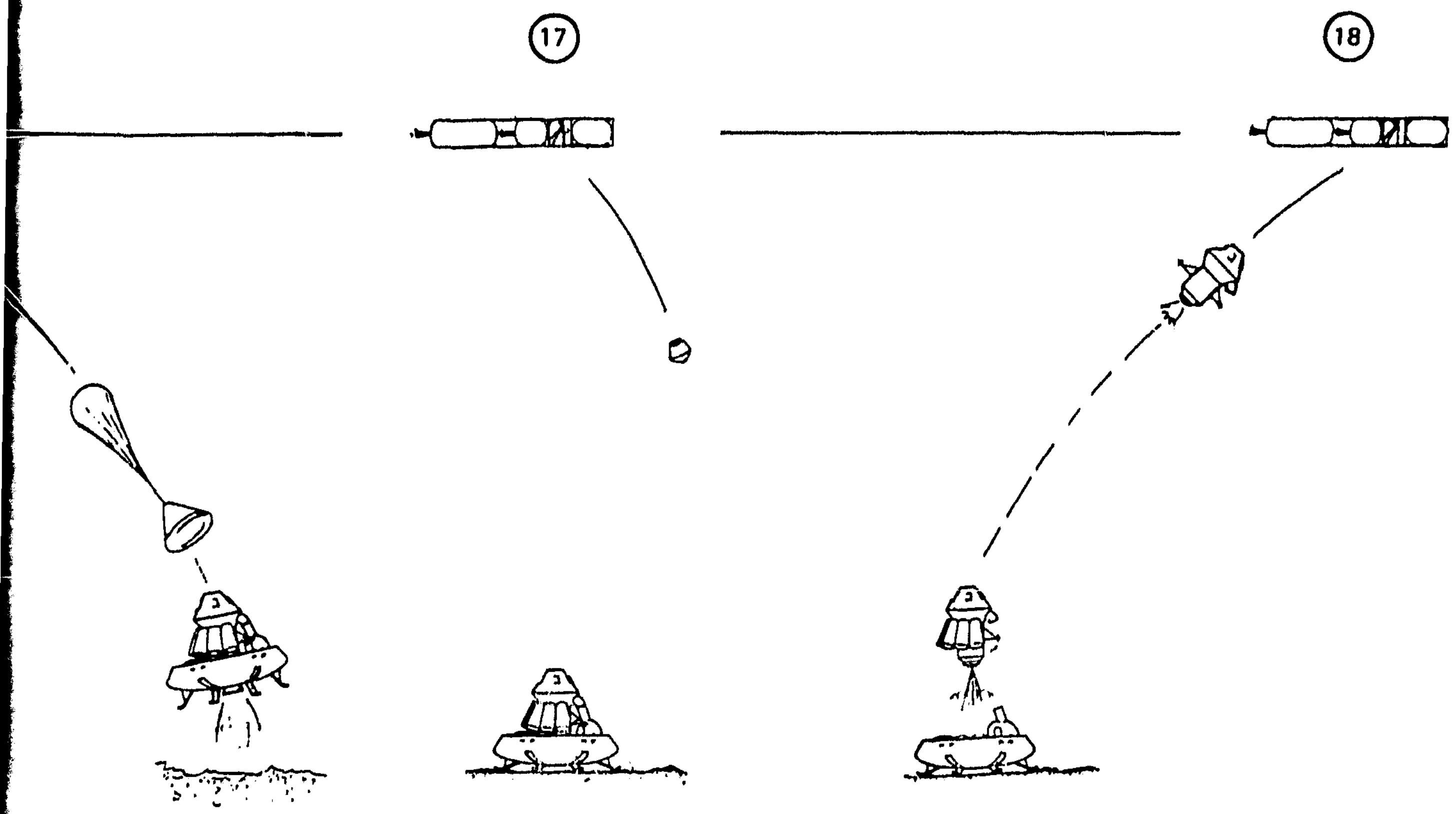
PHASE TIME / CUM TIME	2 TO 5 DAYS		~ 2 HOURS	

⑭	⑮	⑯	
• LAUNCH ENGINEERING PROBES • SURVEY FOR MEM LANDING SITES	JETTISON MEM INTERSTAGE	• DEORBIT MEM • AEROBRAKE USING BALLUTE AND HEAT SHIELD • LAND, USING DESCENT ENGINE	• CONDUC • LAUNCH • COLLEC SAMPLE

Figure 3.6

FOLDOUT FRAME

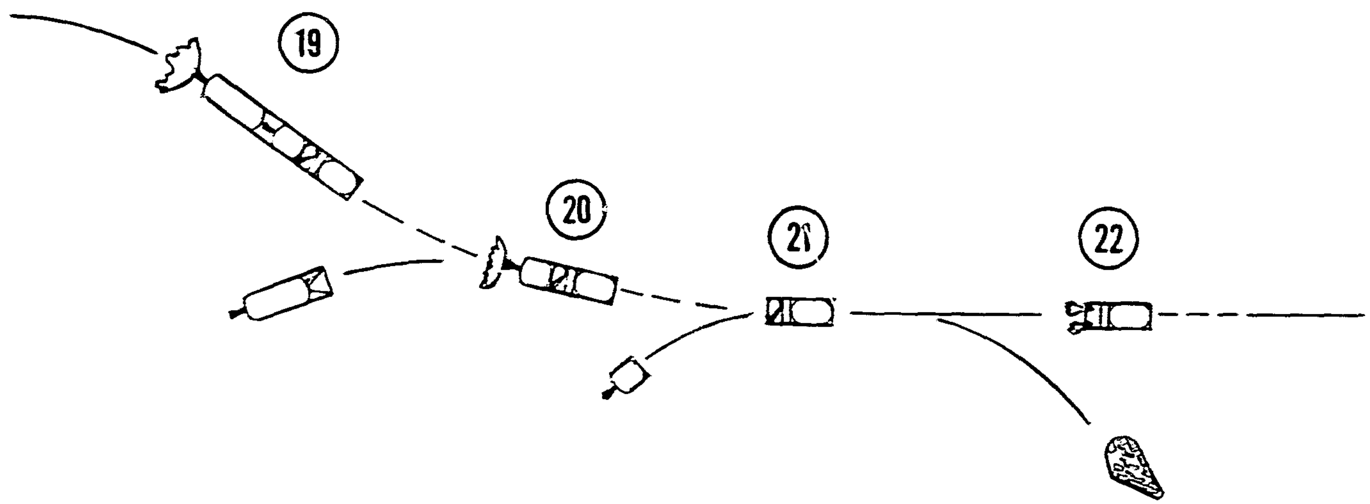
MARS OPERATIONS



OURS	30 DAYS	~ 2 HOURS
16	17	18
EM E USING BALLUTE SHIELD IG DESCENT	• CONDUCT ORBITAL SCIENCE • LAUNCH SCIENTIFIC PROBES • COLLECT SURFACE DATA AND SAMPLES	• MEM TWO-STAGE ASCENT • RENDEZVOUS AND DOCK WITH SPACECRAFT

Figure 3.6-10 Manned Mars Landing Mission-Chemical/Aerobraking/Chemical (Sheet (3)

MARS – EARTH TRANSIT OPERATIONS



PHASE TIME			210-250 DAYS	<1 DAY
CUM TIME				460-550 DAYS (EXCLUDING EARTH ORBIT ASSEMBLY)
	19	20	21	22
	• TEI STAGE 1 BURN	• JETTISON TEI STAGE 1 • COMPLETE TEI USING TEI STAGE 2	• JETTISON TEI STAGE 2 • COAST • THREE MIDCOURSE CORRECTIONS • ANALYZE SURFACE SAMPLES AND EXPERIMENT DATA	• SEPARATE EARTH ENTRY MODULE FROM MISSION MODULE • DIRECT REENTRY RETURN AND WATER RECOVERY OF EEM • DIVERT MISSION MODULE TO NON-EARTH-IMPACT TRAJECTORY

Figure 3.6-10 Manned Mars Landing Mission-Chemical/Aerobraking/Chemical (Sheet 4)

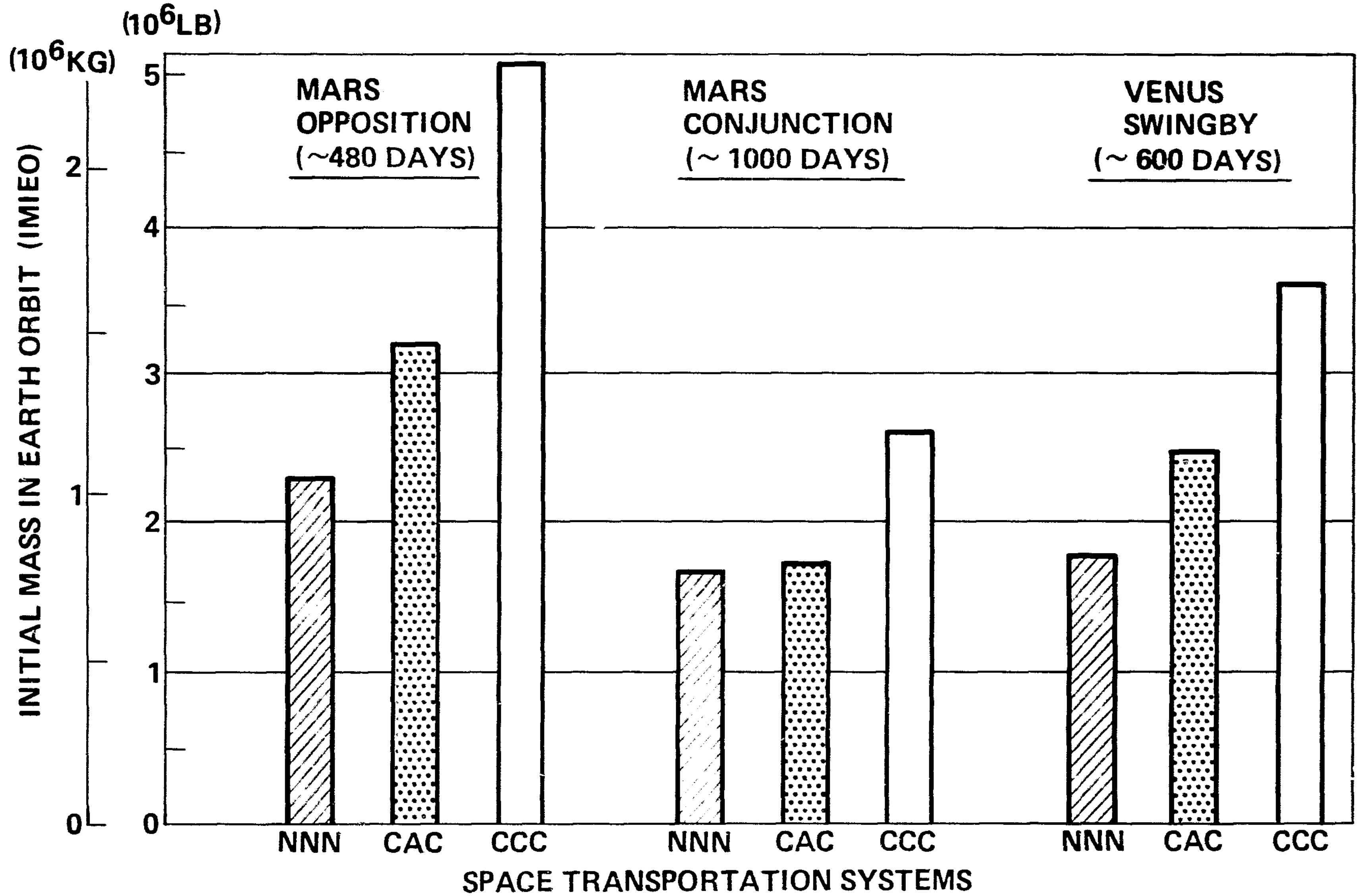


Figure 3.6-11. Manned Mars IMIEO Comparison

Single stage LO_2/LH_2 systems from the GSS and lunar programs appear adaptable for manned planetary missions since their average mass was approximately 272 700 kg (600,000 lbs) resulting in a typical cluster of six modules for TMI and off-loading one module each for the MOI and TEI stages. The nuclear stage sizes for GSS and lunar programs are equally adaptable but the nuclear systems were judged to be of doubtful practicality for those missions.

3.6.1.4.3.3 Earth Launch Requirements Comparison

The number of HLV launches required to deliver the manned planetary space vehicle to Earth orbit is shown in figure 3.6-12. Included in this quantity of launches is that required for the mission spacecraft since in the aerobraking concept this unit is considerably heavier than the mission spacecraft for the all-propulsion modes due to thermal protection. Only the HLLV has been considered for launching the system elements although space shuttle would be used to deliver crews/supplies to orbit.

3.6.1.4.3.4 Operational Comparison

An operational comparison was not developed.

3.6.1.4.3.5 Practicality Assessment

The nuclear system requires development of a unique propulsion technology and system (the unique technology was demonstrated by the ROVER/NERVA nuclear rocket program) that may have no other practical uses. Thus its efficiency relative to the chemical system must be evaluated with due consideration of the potential differences in unique development costs.

The chemical/aerobraking (CAC) system requires dependence on the large aerobraking device; it cannot be practically demonstrated excepting possibly through a subscale automated mission. (The same objection can be raised for the MEM and could have been raised for the Apollo LM).

The all-chemical system requires assembly of a comparatively massive system on orbit but otherwise appears quite practical. Large Earth orbital or cislunar stages are applicable to the trans-Mars injection maneuver if they are designed for multiple clustering, but would require modification to reduce boiloff for the maneuvers at Mars.

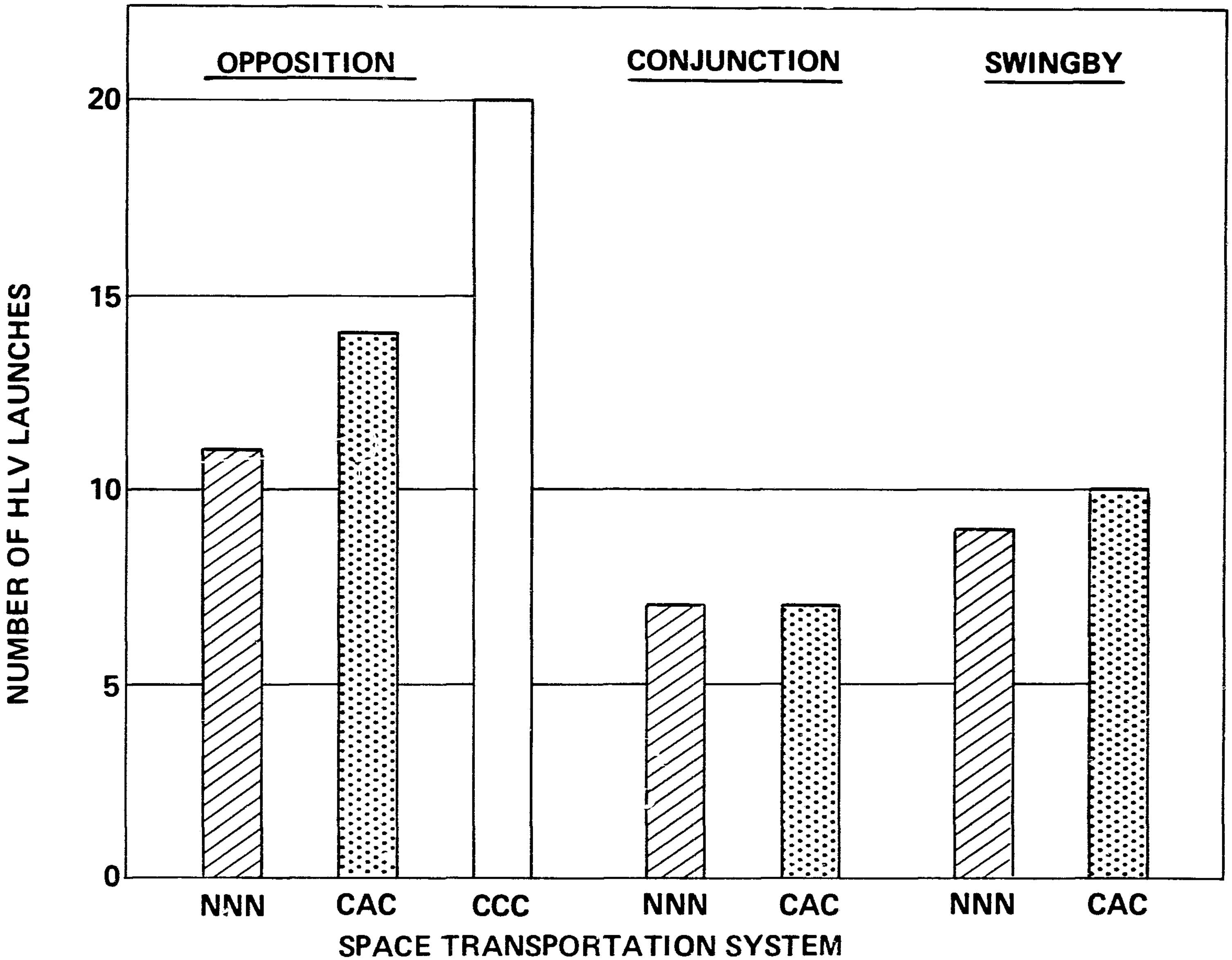


Figure 3.6-12. Earth Launches Required for A Manned Mars Mission

3.7 AUTOMATED LUNAR EXPLORATION PROGRAM

3.7.1 Automated Lunar Missions

3.7.1.1 Mission Summary

The program objective is unmanned exploration of the lunar surface, including high latitude and backside regions, with the following capabilities:

- Surface mobility (long duration traverses)
- Deployed science stations (extension of ALSEPS net)
- Sample return (of material collected on traverses)
- Broadband scientific observation of the total lunar surface.

Figure 3.7-1 shows general operational features of the various program hardware elements. The science satellite is located in polar orbit; it releases a subsatellite into the same orbit. A backside landing and a traverse by a deployed rover are also shown. Communications during backside operations are relayed by a "halo" orbit satellite. Science stations are set out by the rover, which performs various scientific functions including sample collection during the traverse. Samples are brought by the rover to the return system mounted on the lander platform; a departure to Earth by the sample return is shown.

3.7.1.2 Mission Systems Description

3.7.1.2.1 Mission Options

Not applicable.

3.7.1.2.2 Payload Descriptions

Figure 3.7-2 shows the general arrangement of the orbital spacecraft systems used by this program, along with their mass and volumes. These volumes correspond to launch configurations. Antennas, solar arrays, etc., have been folded and stored for launch.

The general characteristics of the lander platform and of the rover and return system it carries are shown in figure 3.7-3. The delivery size shown is for the composite package as prepared for launch. The Earth atmosphere entry return system variant is pictured. Its weight is approximately one-quarter of that of the Earth parking orbit return variant; when that option is used a heavier lander is required.

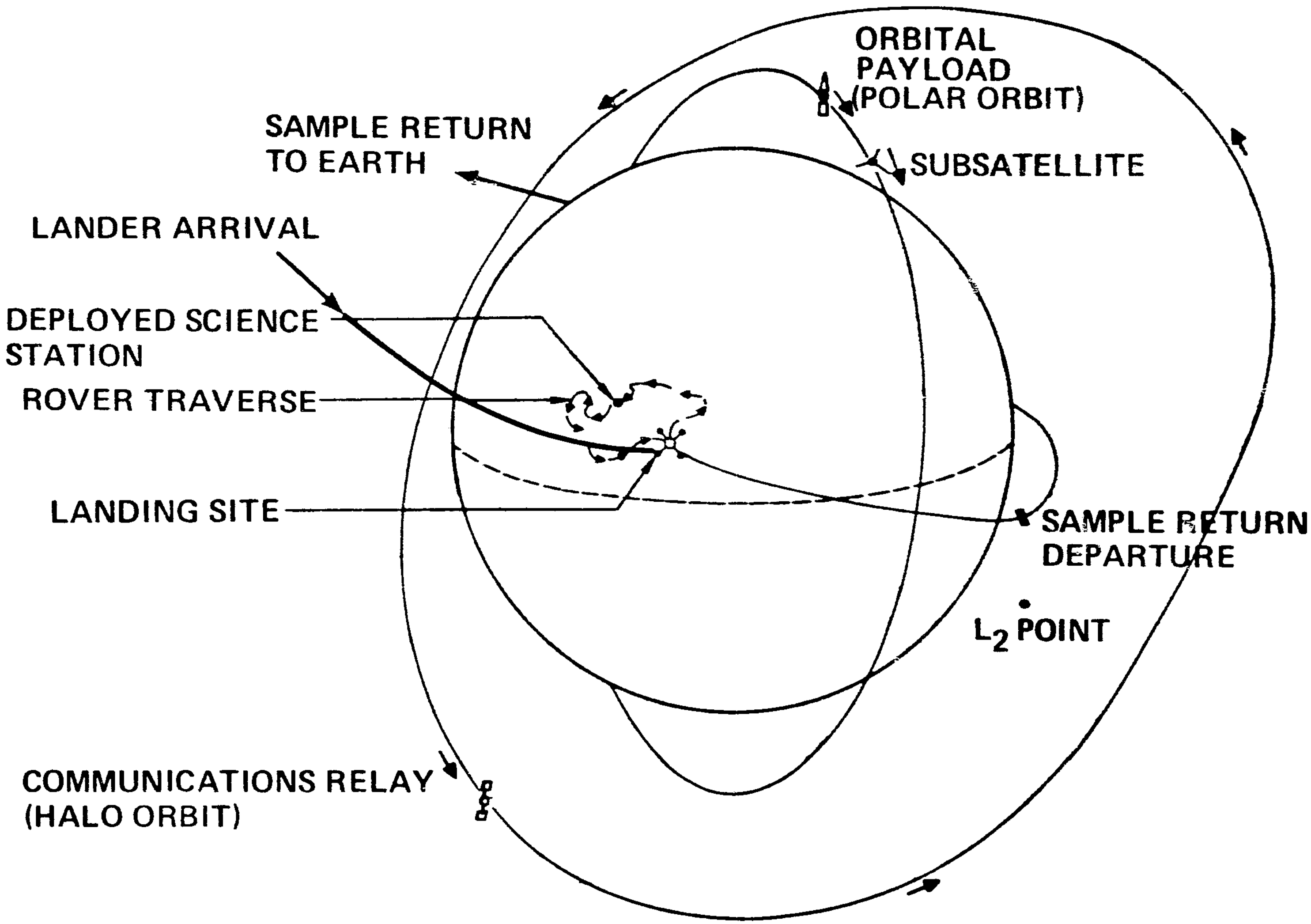
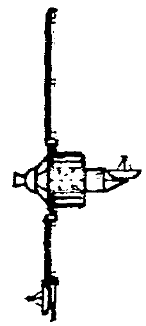
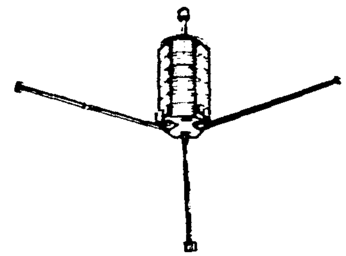
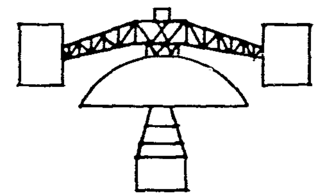


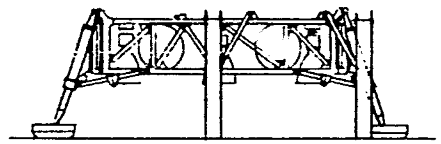
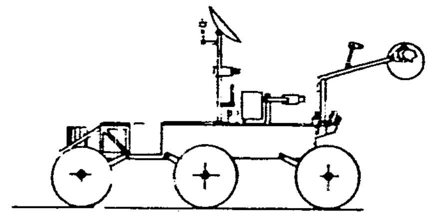
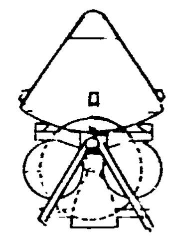
Figure 3.7-1. Automated Lunar Operations

D180-18768-1

	SCIENCE ORBITER	SUBSATELLITE	COMMUNICATION RELAY
CHARACTERISTICS			
FEATURES	• 60 N MI POLAR ORBIT • INSTRUMENTS • OPTICAL SCANNER • MICROWAVE SCANNER • MAGNETOMETERS	• 60 N MI POLAR ORBIT • USED IN CONJUNCTION WITH SCIENCE ORBITER • PARTICLE AND FIELD INSTRUMENTS	• BACK-SIDE HALO ORBIT • MODIFIED ATS-F
DELIVERY MASS* KG (LB)	455 (1,000)	59 (130)	909 (2,000)
	514 (1,130)		
DELIVERY SIZE M (FT) D x L	2.1 x 3 (7 x 10)		3 x 4.6 (10 x 15)

* WITH GROWTH

Figure 3.7-2. Automated Lunar Orbiting Payloads

CHARACTERISTICS	LANDER SYSTEM	ROVER	SAMPLE RETURN SYSTEM
			
FEATURES	• TRANSPORTS ROVER AND SAMPLE RETURN SYSTEM • TERMINAL LUNAR LANDING SYSTEM	• COLLECT SURFACE SAMPLE • DEPLOY SCIENCE STATIONS	• DIRECT EARTH LANDING AND EARTH PARKING ORBIT OPTIONS
DELIVERY MASS* KG (LB)		489 (1,075)	
DIRECT EARTH LANDING	732 (1,610)		411 (905)
EARTH PARKING ORBIT	1,641 (3,680)		1,552 (3,415)
	DIRECT: 1,632 (3,590)	PARKING ORBIT: 3,714 (8,170)	
DELIVERY SIZE M (FT) D x L	DIRECT EARTH LANDING 4.4 x 3.4 (14.5 x 11) EARTH PARKING ORBIT 4.4 x 4 (14.5 x 13)		

*WITH GROWTH

Figure 3.7-3. Automated Lunar Sample Return Payloads

These payloads are designed to be launched to Earth orbit integrated with an upper (delivery) stage and protected from air loads by a payload shroud or equivalent.

3.7.1.2.3 Crew Rotation and Resupply Requirements

Not applicable.

3.7.1.2.4 Transfer and Storage

Not applicable.

3.7.1.2.5 Orbital Assembly, Maintenance, and Modifications

Not applicable.

3.7.1.3 Transportation Requirements

3.7.1.3.1 Payload Delivery Points

Payload delivery points are lunar polar and halo orbits for the orbiters. The landers are designed to be released at 8 km (26,000 ft) altitude above the lunar surface at a relative velocity of 110 m/sec (360 ft/sec) or less.

3.7.1.3.2 Payload Delivery Options

Not applicable.

3.7.1.4 Mission/Transportation Modes and Operations

3.7.1.4.1 Transportation Options

Delta V requirements are as follows:

- **Landers**: Earth departure 3 150 m/sec (10,335 ft/sec); deceleration over lunar surface prior to lander release, 2 950 m/sec (9,678 ft/sec).
- **Polar Orbiter**: Earth departure 3 150 m/sec (10,335 ft/sec); orbit insertion 950 m/sec (3,117 ft/sec).
- **Halo Orbiter**: Earth departure 3 150 m/sec (10,335 ft/sec); lunar flyby 190 m/sec (623 ft/sec); halo orbit insertion 150 m/sec (492 ft/sec).

The polar orbiter is within the capability of shuttle/IUS. The Halo orbiter is within the capability of shuttle and a liquid IUS or of shuttle/solid IUS provided that a small liquid stage is added to the Halo orbiter to perform the two small delta v's in the lunar vicinity. The landers are within the

capability of a shuttle/Full Capacity Tug (FCT) with the tug expended in the case of the larger lander. Advanced transportation systems as analyzed by this study are not required. A small OTV could be used in lieu of IUS or FCT if desired. Transportation analyses were not conducted for the automated lunar mission except to verify the above capabilities.

3.8 AUTOMATED PLANETARY PROGRAM

The current shuttle traffic model for automated payloads contains 14 mission types, including comet flyby, asteroid rendezvous, Saturn orbiters, a small Mars sample return system, and other advanced missions. Rather than attempt to forecast all possible or probable advanced missions of an entire future unmanned program, three missions were selected that are representative of the degree of technology advancement and equipment size that may occur in the time period considered in this study.

The three missions selected as representative of such an advanced program are

- A large Mars sample return system
- A Jupiter Buoyant Probe
- A Ganymede Lander

There are no principal references for these missions, although the sample return mission has many similarities to the shuttle traffic model sample return mission.

3.8.1 MARS SAMPLE RETURN

3.8.1.1 Mission Summary

3.8.1.1.1 General Description

The mission goal is exploration of the Martian surface with return of 10 kg (22 lb) of surface and atmosphere samples.

The illustration shown in figure 3.8-1 is of operations on the Martian surface. The lander stands on its legs with the two rover ramps deployed. One rover is on the ramp; the second is near the sample return system receptacle. This receptacle brings the sample canister to its launch location in the return system which is mounted on the lander. The lander serves as launch pad for the sample return ascent stage.

The lander shown carries two rovers to increase the probability of mission success; a sample can be obtained even if one rover fails or is involved in an accident.

3.8.1.1.2 Mission Assumptions and Constraints

Mission assumptions and constraints are summarized in table 3.8-1.

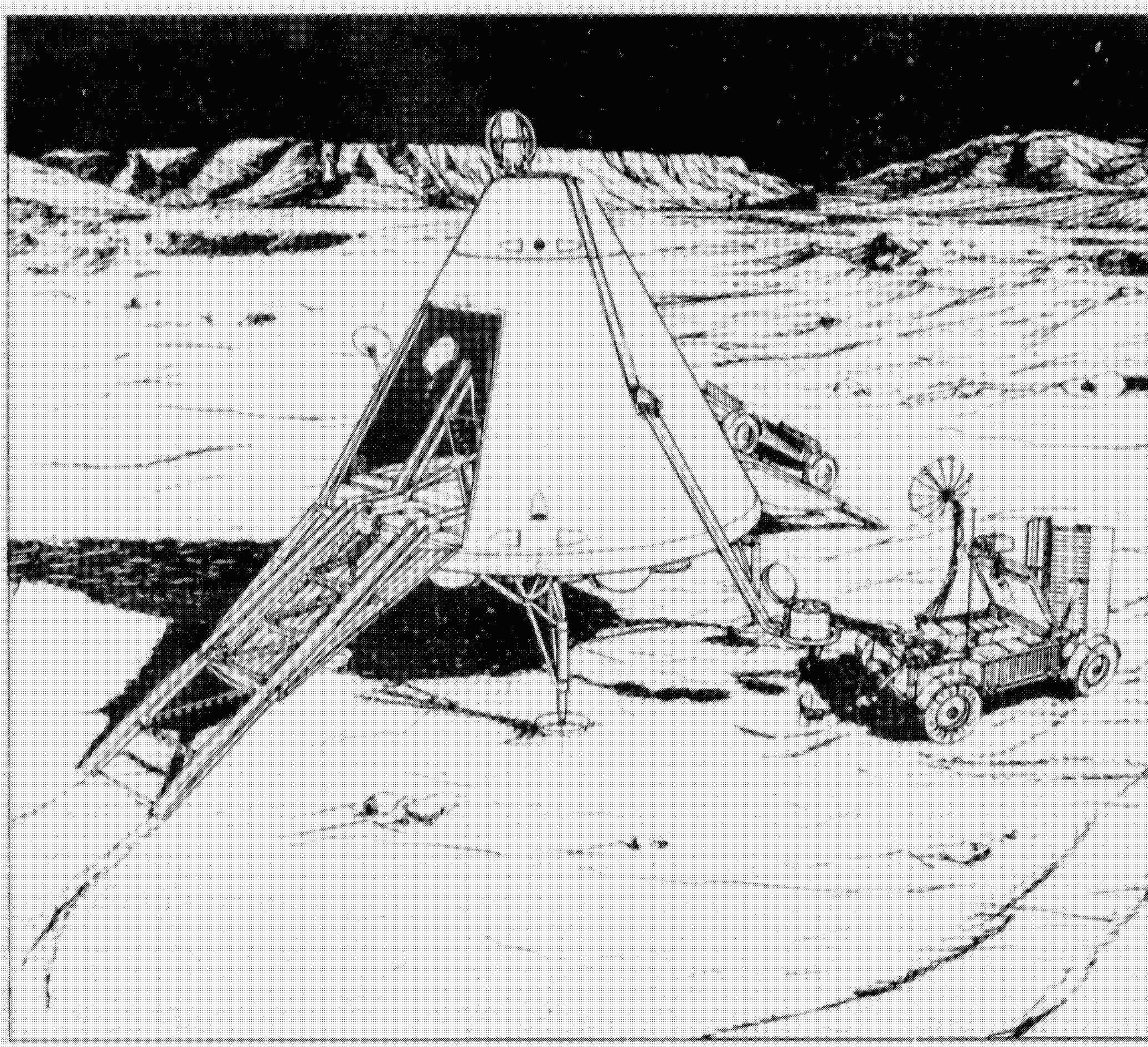


Figure 3.8-1. Rover Places Sample for Loading into Ascent Stage

Table 3.8-1. Mission Assumptions and Constraints

Mission	Objectives	Mission Assumptions and Constraints
Mars Rover with selected sample return	• Investigation of numerous Mars surface features on a continuous traverse, collection of selected sample and return to earth	• On-board avionics to permit semi-autonomous operation • Navigate to selected Martian coordinates with path-adaptive guidance for obstacle avoidance • Communications through relay satellite(s) • Sample pickup for observation; optional return to earth up to 12 kg • Automated biological laboratory • All-weather descent capability, or hold provisions in orbit for suitable weather

3.8.1.2 Mission Systems Description

3.8.1.2.1 Mission Options

Two modes of sample return were considered:

- Direct Earth atmospheric entry by a sealed capsule (note: quarantine requirements may be imposed on this mission.)
- Return to an Earth parking orbit for retrieval by a vehicle such as the space shuttle

Two mission profiles were considered; conjunction (slow) and opposition (fast) profile types. This leads to a total of four mission options.

Figure 3.8-2 shows the four options that result from the basic possibilities of slow or fast mission return and Earth parking orbit or Earth atmospheric entry. Slow return eases the propulsion requirement but increases the operational mission period; this has a possible impact on reliability.

3.8.1.2.2 Payload Description

The sample return propulsion system is here considered as a payload to permit sizing of the lander system and determination of the physical relationships to the rover and the sample handling equipment. The lander descent engines and propellants are included for the same reasons. Propulsion systems were based on Earth-storable propellants, Isp = 3 140 m/sec (320 seconds). Stage propellant mass fraction was varied with size and ranged from 0.84 to 0.90.

3.8.1.2.2.1 Sample Return Systems

Both sample return types will require the ability to perform one or more midcourse maneuvers to

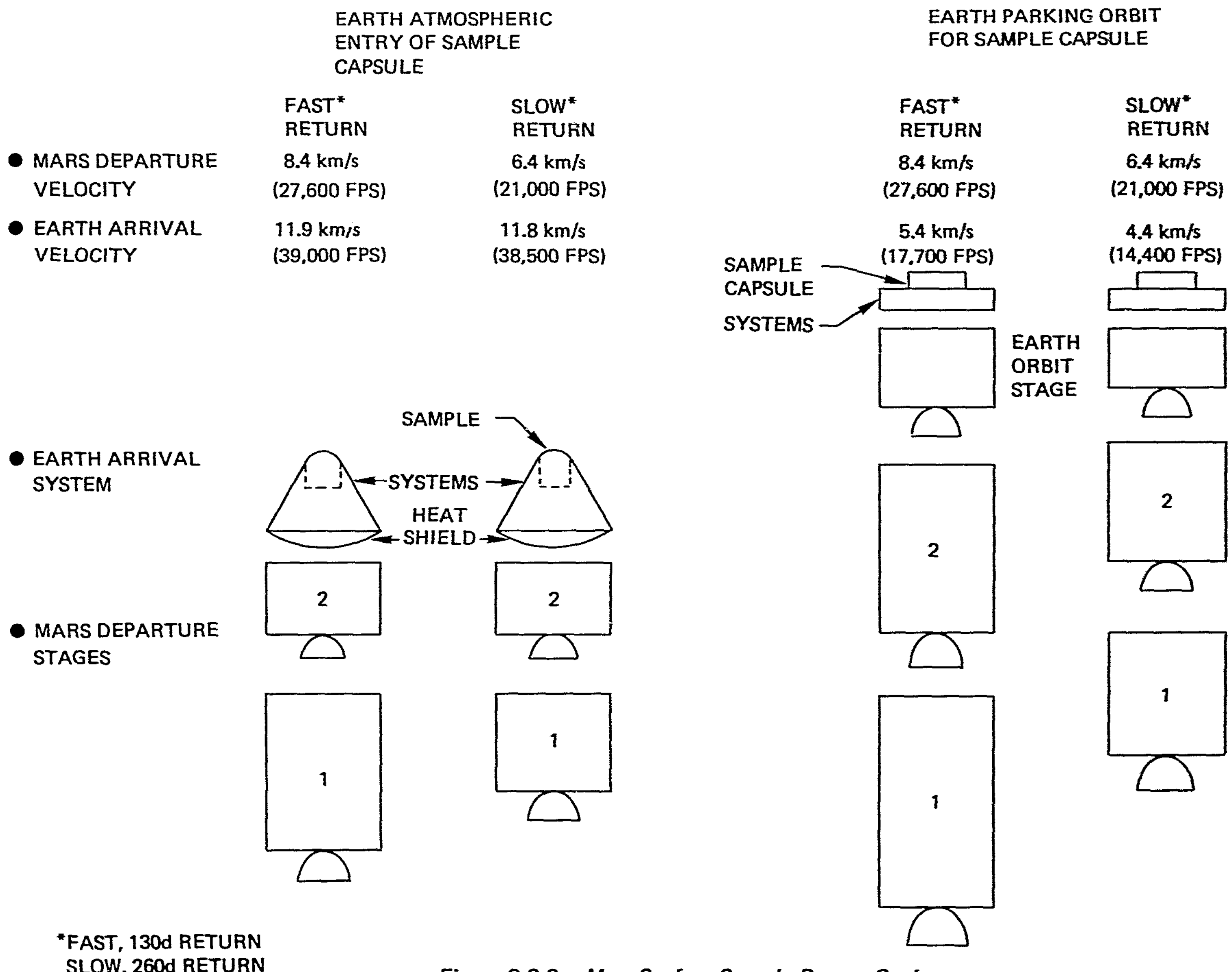


Figure 3.8-2. Mars Surface Sample Return Options

ensure accurate arrival at Earth. Consequently, the returning payload includes the following capabilities:

- Tracking beacon
- Command reception/decode
- Attitude reference system: inertial plus stellar/solar
- Power supply for the above (solar cells plus battery)
- Attitude control
- Start/stop vernier engine capacity
- Environmental control

The Earth-entry option requires a thermal-protection system (heat shield), drogue, and main parachutes, flotation system (even if aerial retrieval is baselined), and recovery aids. Table 3.8-2

Table 3.8-2. Sample Return Systems Weight Summary

COMPONENT	EARTH ATMOSPHERE ENTRY OPTION		EARTH PARKING ORBIT OPTION	
	KG	LB	KG	LB
SAMPLES*	10	22	10	22
CONTAINMENT/CORE TUBES*	2	4	2	4
HATCH/ACTUATOR	1	2	1	2
GUIDANCE/NAV./COMMUNICATIONS	15	33	15	33
ELECTRIC POWER	10	22	10	22
ENVIRONMENTAL CONTROL	5	11	5	11
ATTITUDE CONT./MIDCOURSE	20	44	25	55
AEROSHELL/OTHER STRUCTURE	45**	99	20	44
PARACHUTE/RECOVERY AIDS	32	71	2	4
SYSTEM TOTAL	140	309	90	198

*INCLUDES SAMPLES OF MARTIAN ATMOSPHERE

**HEAT SHIELD WEIGHT ESSENTIALLY CONSTANT FOR RETURN TRIP TIMES OF 100 DAYS OR MORE

summarizes estimated weights for these systems; the Earth parking orbit option systems are integrated with the propulsion system shown in figure 3.8-2 as the "Earth orbit stage."

3.8.1.2.2.2 **Rover Collection Systems**

A rover system was configured capable of long range traverses and sample collection from a range of Martian surface features. Since communication (Mars/Earth/Mars) will always be at least 8 minutes, the rover is semi-autonomous, primarily in the area of obstacle/hazard recognition and avoidance. This rover is illustrated in figure 3.8-3; a weight statement is given in table 3.8-3.

Table 3.8-3. Automated Surface Vehicle (Rover) Mass Statement

Item	Mass	
	kg	lb
Mobility and structure	171	377
Power (RTG)	259	571
G&N	31	68
Communications	36	79
Data	31	68
Science	81	179
Total	609	1342

3.8.1.2.2.3 **Mars Landing System**

The Martian atmosphere (surface pressure $\approx 10^3 n/M^2 \approx 10$ mb; scale height ≈ 11 km $= 3.6 \times 10^4$ ft) allows parachute deceleration of landing payloads. Supersonic drogues and extremely large main parachutes are involved. The Viking System (1 000 kg [2,200 lb]) is probably representative of a maximum practical parachute system. Landing weights for this program are as high as 10 000 kg (22,000 lb); therefore, parachutes are not assumed to be practical. For a planform loading of 300 to 400 kg/M^2 (60 to 80 lb/ft^2) and a drag coefficient of 0.9, terminal descent velocity of an "Apollo shape" will be approximately 500 m/s (1,100 ft/sec), a reasonable range for rocket deceleration. Figure 3.8-4 is an all-inboard profile of such a lander; it incorporates a three-stage ascent system for the Earth orbit return option.

During the final descent after the entry heat pulse, the base heat shield is jettisoned, exposing the nozzles of the landing engines, and the landing gear legs are deployed. The landing engines, throttling as required, use inputs from a Doppler radar and flight control system to bring the lander to a soft touchdown. The lander serves as launch pad for the ascent stage.

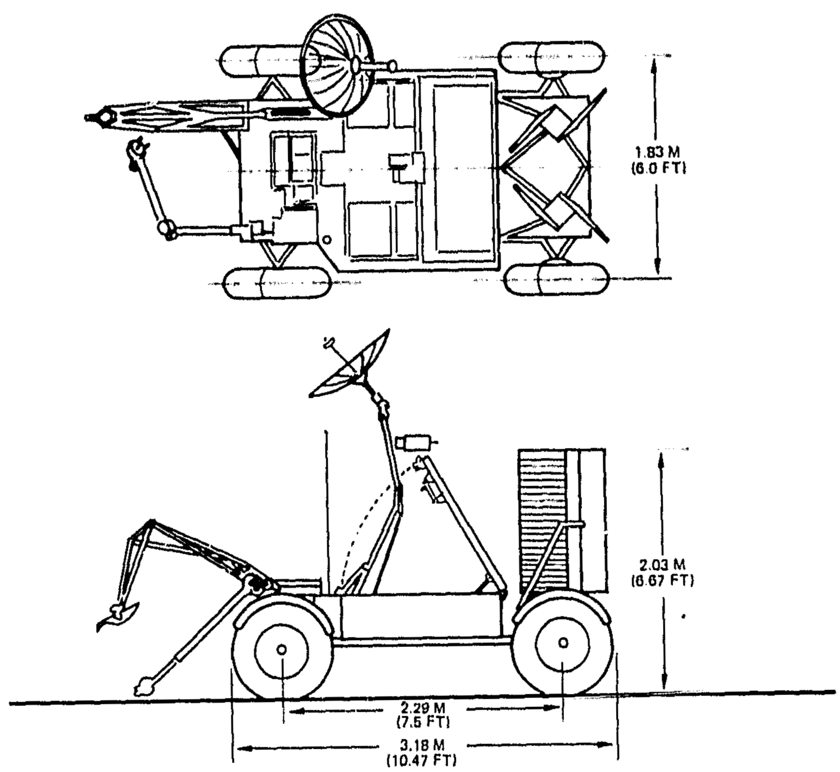


Figure 3.8-3. Mars Rover Vehicle

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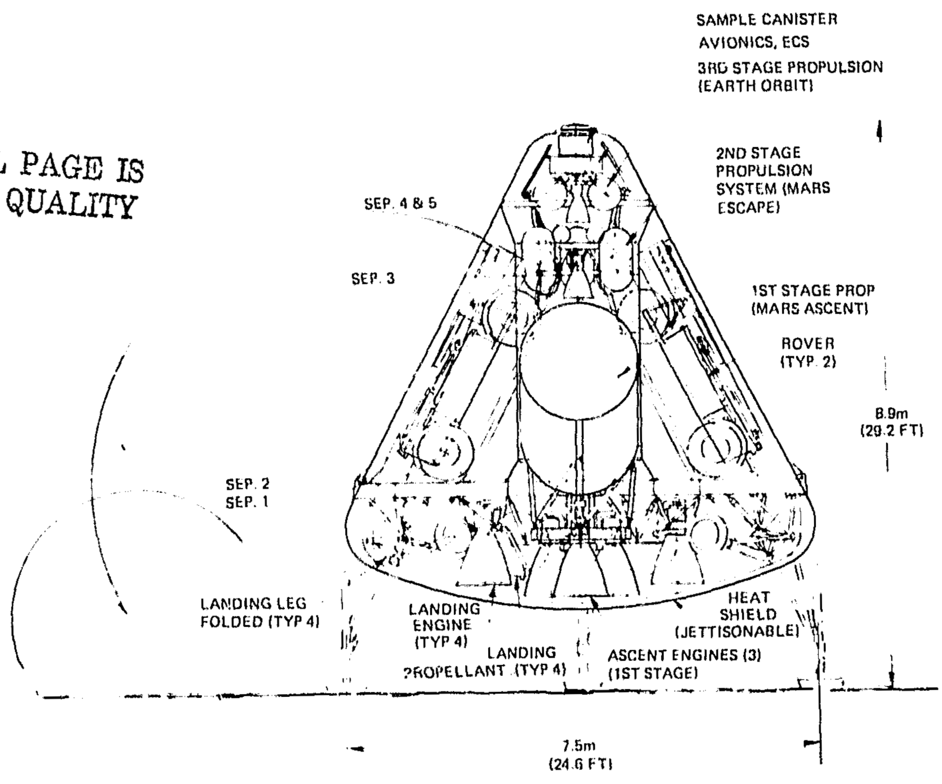

Figure 3.8-4. Mars Sample Return System

3.8.1.2.2.4 Mass Summary

Table 3.8-4 is a mass summary of the lander including all payload items.

Table 3.8-4. MSR Mass Summary

Item	Mass kg	Mass lb
Sample system	90	198
Earth parking orbit stage	788	1,737
Ascent vehicle system	13370	29,475
Rovers (2)	1218	2,685
Lander inerts	5625	12,400
Lander propellants	5300	11,684
Total	26391	58,179

3.8.1.2.2.5 Comparison of Options

The four mission options described result in a wide range of lander (payload) masses and sizes. These are compared in table 3.8-5. Mass values do not include growth allowances. The Earth parking orbit return, fast mission option resulted in an excessive mass and was dropped form further consideration.

Table 3.8-5. Masses of Sample Return Options

Component	Earth Atmosphere Return Option				Earth Parking Orbit Return			
	Fast Return		Slow Return		Fast Return		Slow Return [1]	
	kg	lb	kg	lb	kg	lb	kg	lb
Sample system	140	310	140	310	90	200	90	200
Earth orbit system	–	–	–	–	1488	3,280	788	1,737
Mars escape system	10920	24,075	2440	5,350	51685	113,945	13370	29,475
Rovers	1218	2,685	1218	2,685	1218	2,685	1218	2,685
Lander system	7195	15,860	3212	7,080	42120	92,860	10925	24,085
Total at Mars entry	19473	42,930	7010	15,455	96600	212,970	26390	58,182
Lander dia, m (feet)	6.4	21	4.4	14.5	14.3	47	7.5	24.6

[1] Baseline system, described in text

3.8.1.2.2.6 Crew Transfer and Resupply

Not applicable.

3.8.1.2.2.7 Pickup Points and Transportation Constraints

These payloads are designed to be launched integrated with a delivery propulsion stage and protected from Earth ascent aerodynamic loads.

3.8.1.2.3 Transfer and Storage

Not applicable.

3.8.1.2.4 Orbital Assembly Maintenance and Modification

Not applicable.

3.8.1.3 Transportation Requirements

3.8.1.3.1 Payload Delivery Points

These payloads require delivery to a circular Mars orbit at 1 000 km (540 nmi) altitude.

3.8.1.3.2 Payload Delivery Options

The four options described are shown in table 3.8-6. Mass values include a 20 percent growth allowance on hardware masses.

The heaviest option shown is for a lander larger than some proposed for manned surface exploration, and was not analyzed further. The systems shown here provide for the total return to Earth. The lightest option is sizewise compatible with shuttle launch; the others are not.

Mass and delta V requirements for delivery to Mars orbit are summarized in table 3.8-7. The payload defined above included the sample return transportation.

3.8.1.3.3 Operational Constraints

Operational constraints were not specifically investigated. Principal ones are the infrequent launch opportunities, approximately 26-month intervals, and trajectory and procedural constraints resulting from planetary quarantine requirements.

3.8.1.4 Mission/Transportation Modes and Operations

3.8.1.4.1 Transportation Options

LO_2/LH_2 single stage OTV's were assumed for Earth departure, and LO_2/MMH single-stage OTV's for Mars arrival.

Table 3.8-6. Mars Surface Return Options

CHARACTERISTICS	DIRECT EARTH LANDING RETURN		EARTH PARKING ORBIT RETURN	
	FAST	SLOW	FAST	SLOW
VELOCITIES — MPS (FPS)				
MARS ASCENT & DEPARTURE	8 415 (27,600)	6 402 (21,000)	8 415 (27,600)	6 402 (21,000)
EARTH ARRIVAL	11 890 (39,000)	11 800 (38,700)	5 396 (17,700)	4 390 (14,400)
PAYLOAD MASS * KG (LB)				
@ MARS ENTRY	20 909 (46,000)	5 636 (12,400)	104 545 (230,000)	28 455 (62,600)
PAYLOAD SIZE — M (FT)				
@ MARS ENTRY (D x L)	6.4 x 6.4 (21 x 21)	3.4 x 3.4 (11 x 11)	14.3 x 14.3 (47 x 47)	7.6 x 7.6 (25 x 25)

*WITH GROWTH

Table 3.8-7. MSR Transportation Requirement Summary

Option	Earth Return Mode	Mission	Lander Mass		Earth Departure ΔV		Mars Arrival ΔV	
			kg	lb	m/sec	ft/sec	m/sec	ft/sec
1	Atmos. entry	Fast	20 700	45,540	5 000	16,400	5 400	17,700
2	Atmos. entry	Slow	7 500	16,500	4 300	14,104	2 200	7,200
3	Earth parking orbit	Slow	18 400	62,480	4 300	14,104	2 200	7,200

3.8.1.4.2 Representative Transportation Systems

All arrival stages use dense propellants, LO_2/MMH, with Isp = 3 630 m/sec (370 sec). An allowance of 100 m/s (320 ft/sec) for midcourse capability is provided for this stage. Earth departure stages use LO_2/LH_2, Isp = 4 500 m/sec (459 sec). Mass fractions are based on the studies of orbit transfer vehicles described in Section 5. Mass data and stage sizing are summarized in table 3.8-8.

Table 3.8-8. Mars Sample Return Mass History Summary

System Element	Option 1		Option 2		Option 3	
	kg	lb	kg	lb	kg	lb
Mars Lander (Mission payload)	20 700	45,540	7 500	16,500	28 400	62,480
Arrival stage inerts	7 965	17,560	960	2,120	2 900	6,390
Arrival stage propellant	98 235	216,600	7 040	15,520	26 100	57,540
Arrival stage total	106200	234,160	8 000	17,640	29 000	63,930
Arrival total	126 900	279,700	15 500	34,140	57 400	126,410
Departure stage inerts	27 300	60,200	4 350	9,590	12 400	27,300
Departure stage propellant	314 300	692,900	31 700	69,900	111 700	246,300
Departure stage total	341 600	753,100	36 050	79,490	124 100	273,600
Total initial mass	468 500	1,032,800	51 500	113,630	181 500	400,000

3.8.1.4.3 Transportation Options Comparison and Evaluation

Other alternatives were not analyzed. The three mission options described are all practical in the sense of being within the capability of transportation systems comparable to those required for the manned geosynchronous and lunar missions. However, the practicality or desirability of automated

planetary missions requiring multiple HLLV launches is at least dubious. One of the MSSR options was well within the capability of a single HLLV launch. The idea of a MSSR mission in a single HLLV launch, capable of extended automated surface traverse and sample selection, significant (e.g., 10 kg; 22 lb) sample return, no requirement for Mars orbit rendezvous, and comparatively free from stringent mass limits, appears attractive based on the limited examination accomplished in this study.

A comparison across the three automated planetary missions is provided in paragraph 3.8.4.

3.8.2 JUPITER BUOYANT PROBE

3.8.2.1 Mission Summary

3.8.2.1.1 General Description

This is the second of the three automated planetary missions. Mission objectives are the acquisition of data from the region of the upper Jovian cloud system over one day-night period (10 hours), with further one-time data acquisition to depths of at least 500 km (170 n.mi.) below the cloud tops.

Long duration mission data is collected by a science platform supported by a balloon. In the illustration (figure 3.8-5) it is just above the cloud tops; the moon Io is visible. The deep probe has just been released and will descend into regions of increasing pressure and temperature. It relays data to the buoyant probe, for transmission to an orbital relay and thence to Earth.

It is possible that the Jovian atmosphere is too violent (shears and gusts) to allow operation of a balloon. Alternative mechanizations might be visualized. Clearly, additional data on the planet are needed to allow firm selection of a design approach.

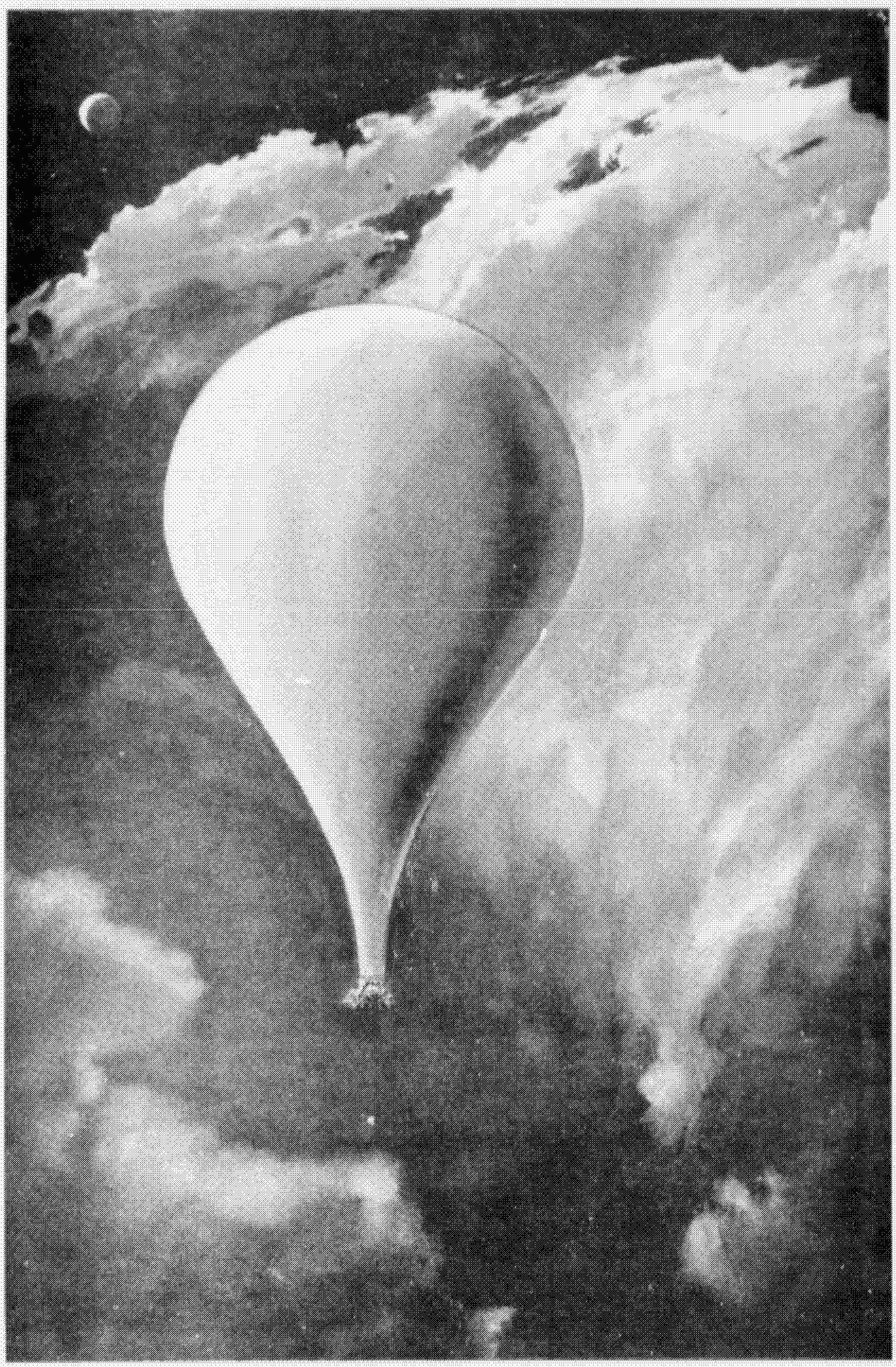

Figure 3.8.5. Buoyant System Releases Deep Probe

3.8.2.1.2 Mission Assumptions and Constraints

Mission assumptions and constraints are summarized in table 3.8-9.

Table 3.8-9. Jupiter Buoyant Probe Mission Assumptions

Mission	Objective	Assumptions and Constraints
Jupiter orbiter/ probe	• Investigate conditions in Jovian ionosphere and atmosphere at various depths	• Basic probe principle of operations to be that of a buoyant probe which can be positioned at various levels in atmosphere. • Probe to release a "free falling" hardened descent probe to return temperature/ pressure measurements from up to 3. • Probe to be released from an atmoshere entry probe. • Probe/entry mission supported by relay orbiter • Buoyant probe to operate for at least one Jovian day/night period • TY = 1980

3.8.2.2 Mission Systems Description

3.8.2.2.1 Mission Options

Not applicable.

3.8.2.2.2 Payload Descriptions

Mission Concepts—The basic principles of this mission are as follows:

- An orbital communications relay satellite is established in orbit around Jupiter; orbital science systems are included in its equipment complement.
- A probe enters the Jovian atmosphere and—after deceleration—descends by parachutes at a velocity and to an altitude where deployment of a supporting balloon system can take place. Scientific investigations take place from this buoyant platform for a period of at least 10 hours; data are relayed to Earth through the orbital system.
- A deep descent probe is released from the buoyant system to gather information from depths up to 500 km (270 nmi) below the buoyant probe. Data are relayed through the buoyant probe and the orbital system.

Buoyant System—The balloon portion of the buoyant system was illustrated in Figure 3.8-5. When inflated the balloon is 30m (100 ft) in diameter. The balloon is filled with ambient air that is heated in the heat exchanger shown. The heat source is the hydrogen of the Jovian atmosphere (82% hydrogen by weight). This is burned with liquid oxygen brought from Earth in the large tank shown within the probe. Combustion products are not allowed to enter the balloon. The aeroshell and aft fairings are jettisoned before balloon inflation while descent on the main parachute (26m (85 ft) diameter) is in process.

Orbital System—Figure 3.8-6 shows the orbital element with the buoyant probe (in its entry aeroshell) attached. Transportation required for attainment of Jupiter orbit is not included. The main communications antenna diameter is 3m (10 ft). Power is drawn from an RTG assembly.

Operational Sequence—Figure 3.8-7 shows mission events versus time. Initial heating of the balloon requires approximately 185 kg (408 lb) of LO_2; thereafter 45 kg (100 lb) are consumed per hour. As the LO_2 is burned off, the system becomes lighter and moves to a higher altitude. Dropping of the deep probe could be a real-time decision; it is shown as occurring at the beginning of the second Jovian day. Reduction in vehicle weight allows it to rise above the cloud tops.

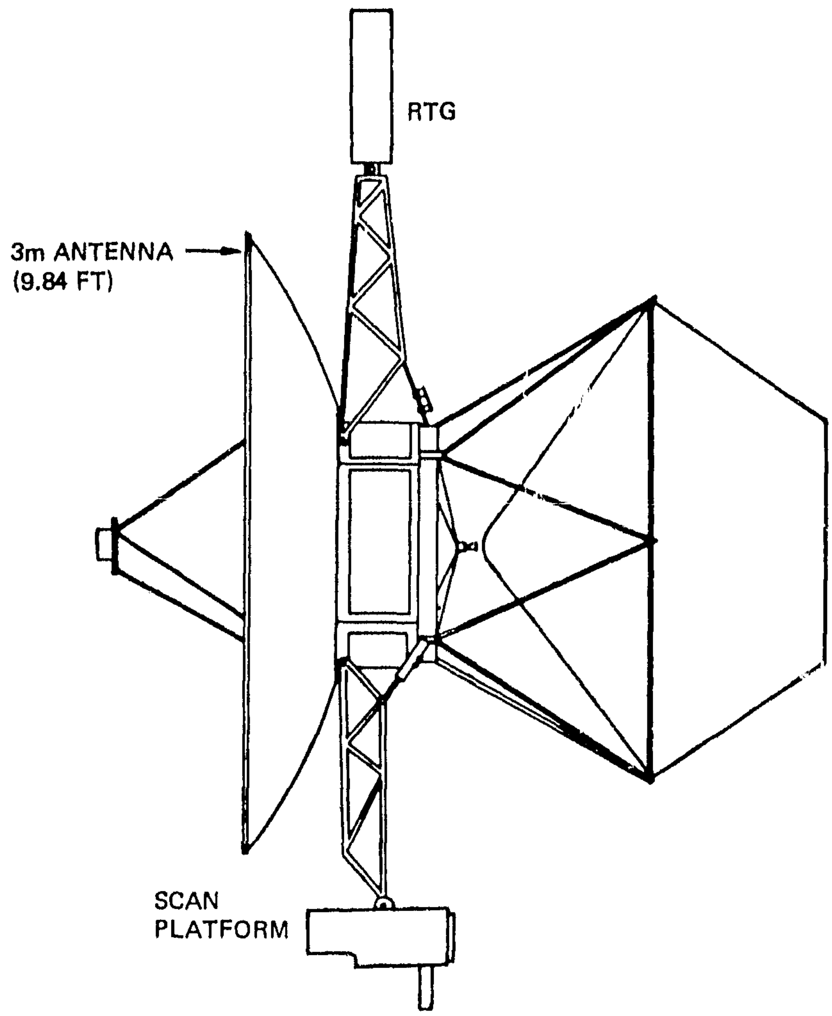

Figure 3.8-6: Jupiter Systems

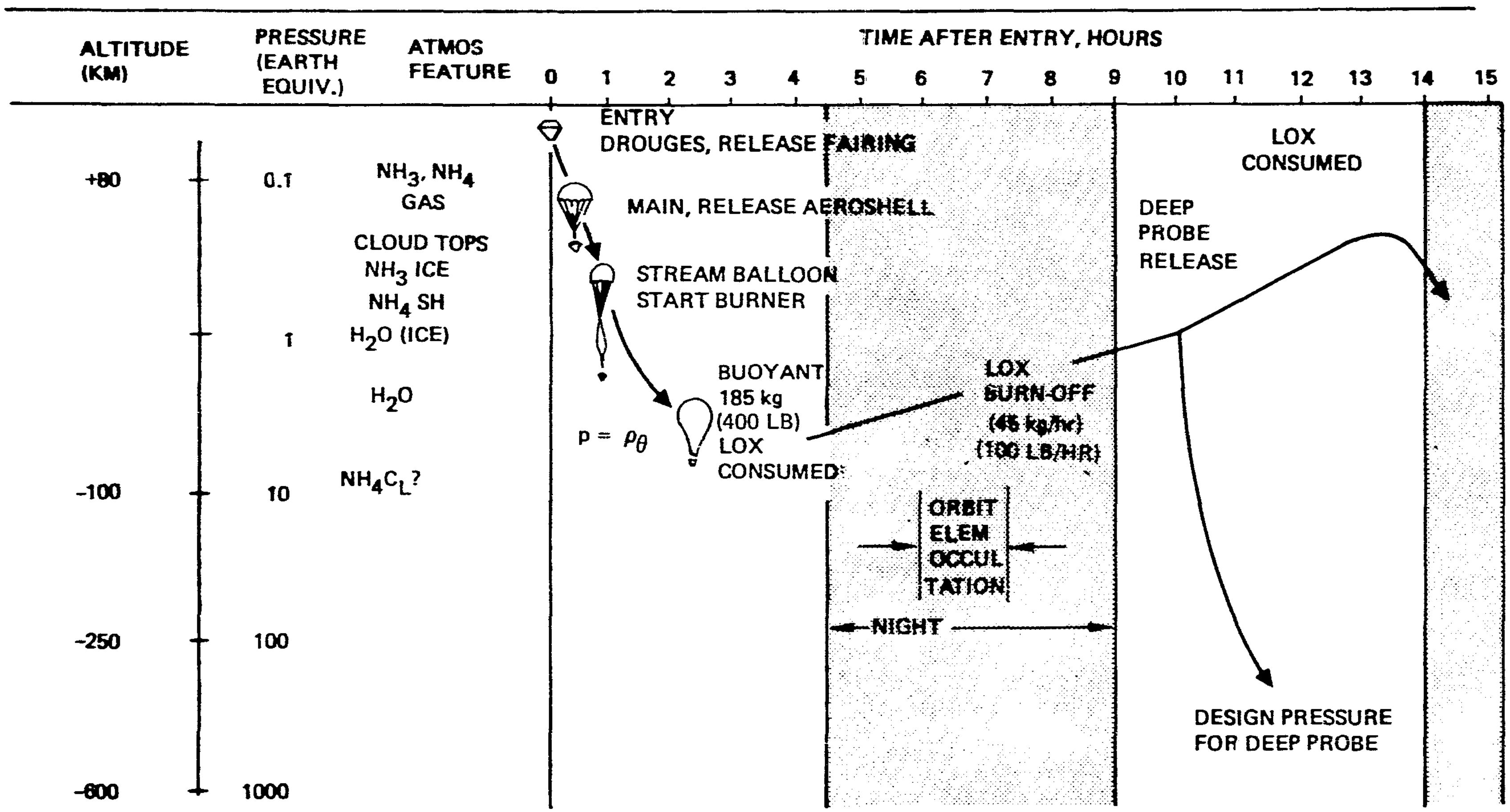

Figure 3.8-7: Jupiter Buoyant Entry Probe Events/Timelines

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Crew Rotation and Resupply Not applicable.

Mass Summary Table 3.8-10 is a mass summary. Note that mass growth allowances are included.

Pickup Points and Transportation Constraints This payload is designed to be launched integrated with a propulsion stage for Jupiter orbit insertion, and protected from Earth ascent aerodynamic loads.

3.8.2.2.3 Transfer and Storage

Not applicable.

3.8.2.2.4 Orbital Assembly, Maintenance, and Modification

Not applicable.

3.8.2.3 Transportation Requirements

3.8.2.3.1 Payload Delivery Options

The Jupiter buoyant probe requires delivery to an elliptical Jupiter synchronous orbit with a period of 9.92 hours. Circular orbits were considered but required extreme delta V's.

3.8.2.3.2 Payload Delivery Options

Not applicable.

3.8.2.3.3 Operational Constraints

Transfer opportunities for Jupiter missions occur at approximately 13-month intervals and last about 20 days.

3.8.2.4 Mission/Transportation Modes and Operations

3.8.2.4.1 Transportation Options

Alternatives were not investigated.

3.8.2.4.2 Representative Transportation Mode and System

3.8.2.4.2.1 Transportation Sequence

The delivery includes an Earth departure with a C_3 of at least 80 km^2/sec^2 (6.89x10^{10}ft^2/sec^2) requiring a delta V of about 6 500 m/sec (21,325 ft/sec) and injection into the Jupiter orbit with a delta V requirement of 9 110 m/sec (29,900 ft/sec). Finally the atmosphere probe must be deorbited with a delta V of 600 m/sec (1,980 ft/sec). The transit time to Jupiter associated with the above C_3 is about 800 to 900 days, depending on the particular mission opportunity.

Table 3.8-10. Jupiter Buoyant Probe Weight Summary

	KG	LB
DEEP PROBE		
SCIENCE/COMMUNICATION	13	29
STRUCTURE	38	84
TPS	12	26
ANTENNAS	1	2
SUBTOTAL	65	145
BUOYANT PROBE		
SCIENCE/COMMUNICATION	40	88
ELECTRICAL POWER (BAT)	40	88
STRUCTURE	80	176
TPS	30	66
PARACHUTE	70	154
BALLOON/INFLATION SYSTEM	130	287
ANTENNAS	2	4
LO_2 TANK	120	265
LO_2	720	1,587
MISC./UNKNOWNS (20%)	73	161
SUBTOTAL	1305	2,877
ENTRY SYSTEM		
AEROSHELL/FAIRING	306	675
COMMUNICATIONS, ANTENNA	19	42
RADIATION SHIELDING	110	243
SUBTOTAL	435	960
ORBITAL ELEMENT		
SCIENCE/COMMUNICATION/ANTENNA	199	439
ELECTRICAL POWER (RTG)	166	366
STRUCTURE	119	262
VERNIER PROPULSION	43	95
MISC./UNKNOWNS	78	172
SUBTOTAL	605	1,334
TOTAL (NO PROPULSION)	2410	5,312

3.8.2.4.2.2 Transportation Sizing

The ΔV's for this mission are extremely demanding. To obtain a "reasonable" Earth departure mass a high performance, space-storable propellant combination was selected: diborane (B_2H_6) and oxygen difluoride. Isp is predicted as 4 170 m/sec (425 sec).

Two stages are required for the above maneuvers. The first stage proves 5.2 km/sec (17,056 ft/sec); the second provides 3.9 km/sec (12,742 ft/sec) for the probe/lander combination, and then 0.60 km/sec (1,968 ft/sec) for the lander deorbit. The velocity split between the stages is chosen to yield minimum mass for the total (table 3.8-11). Propellant mass fractions are appropriate to a pressure-fed system. Some avionic functions are assumed to be provided by the orbiter.

This mass is in the large OTV class as regards the Earth departure maneuver. A C_3 of 80 km^2/sec^2 (6.89×10^{10} ft^2/sec^2) (trip times of approximately 900 days) will require a start of weight of approximately 350 000 kg (770,000 lb) indicating a large OTV mass of 286 000 kg (630,000 lb).

Table 3.8-11 Mass Summary for Jupiter Probe Orbit Insertion System

Stage One (Thrust = 150 000 N (33,721 lb_f))	Mass	
	kg	lb
Inert	7 800	17,160
Propellant	45 960	101,112
Stage	53,760	118,272
Stage Two (Thrust = 30 000 N (6,744 lb_f))	kg	lb
Inert	1 300	2,860
Deorbit propellant	490	1,078
Main burn propellant	6 530	14,366
Stage	8 320	18,304
Payload	2 420	5,324
Total	65 500	141,900

3.8.2.4.2.3 Operational Factors

Operational factors were not analyzed; no significant problems are apparent except that launching OF_2/B_2H_6 propellants in the shuttle payload bay appears undesirable from the risk standpoint.

3.8.2.4.2.4 Earth Launch Requirements Summary

Sixteen shuttle launches or four HLLV launches are required.

3.8.2.4.3 Transportation Options Comparison and Evaluation

This mission appears practical but requires a large and presumably expensive transportation system. The mission appears quite desirable from the science standpoint. The large mass of this mission results from the high orbit insertion delta V at Jupiter. Multiple-pass aerobraking, even if it took several months, might provide a way to reduce the overall mass requirement by as much as a factor of 10. The mission could then be done with two shuttle launches.

3.8.3 GANYMEDE LANDER

3.8.3.1 Mission Summary

This is the third of three automated planetary missions. It was added at the request of NASA Headquarters following the midterm briefing. Mission goals are:

1) Orbital observation of Ganymede surface

2) Soft landing, with sample collection and analysis, stereo TV, plasma physics, magnetometer, seismic instruments.

Orbital maneuvers near Jupiter are based on "Exploring Jupiter and its Satellites with an Orbiter", J. C. Beckman, J. R. Hyde, and S. [illegible] Astronautics and Aeronautics, September 1974.

The basic mission plan is to enter Jupiter space following a flight from Earth of approximately 800 days (C_3 of 80 $km^2/sec^2 = 6.89x10^{10} ft^2/sec^2$). A Ganymede powered swing-by maneuver is used to reduce the ΔV required at Jupiter arrival. A series of Ganymede fly-bys follow (Ganymede pumps) reducing the orbital period (Figure 3.8-8). Four fly-bys take place before the landing; more could be used, but with diminishing return. A maneuver summary is given in table 3.8-12.

Table 3.8-12 Orbit Maneuvers

Day	Maneuver	Spacecraft ΔV, M/S (FPS)	Effective ΔV Savings M/S (FPS)	Resulting Perijove/ Apojove (Jupiter Radii)	Orbit Period Days
0	Jupiter orbit Insertion (powered fly-by)	1 190 (3,900 ft/sec)	500 (1,640)	14.0/166	105
105	Ganymede pump	0	500 (1,640)	13.4/87	43
148	Ganymede pump	0	350 (1,148)	12.9/62	29
177	Ganymede pump	0	300 (984)	12.5/52	21
198	Landing	4 080 0 (13,400 ft/sec)		15.0/15.0*	**

* Ganymede is at 15.0 R_j ** Orbiter is now in 14.4 day orbit, Ganymede period is 7.2 days.

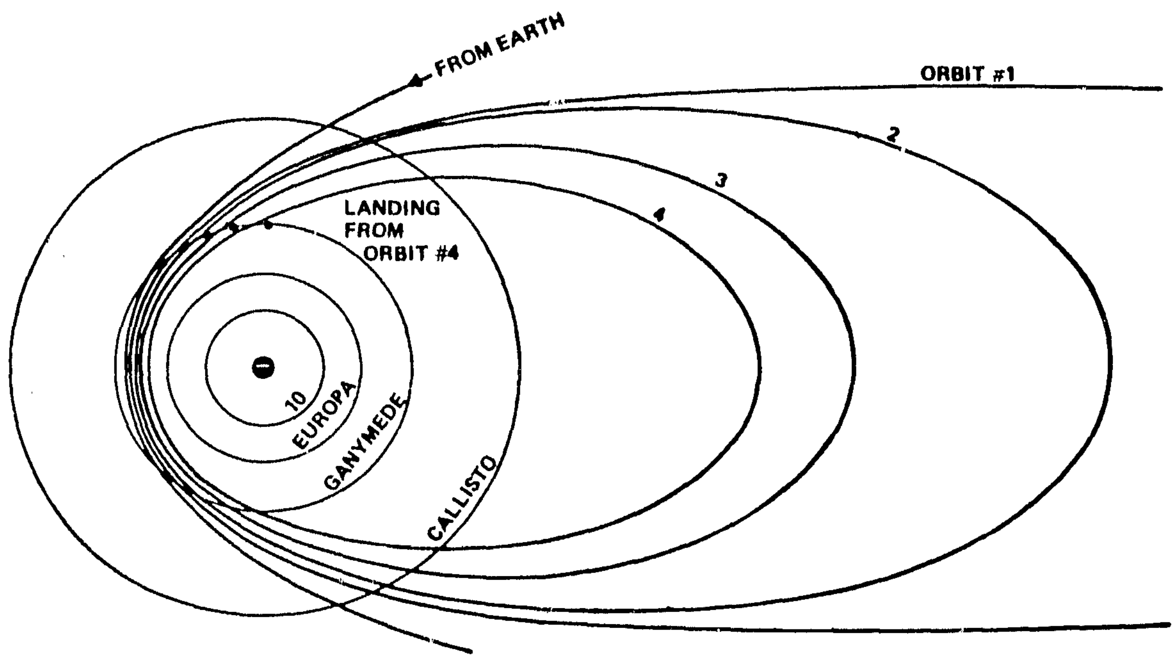


Figure 3.8-8: Orbital Maneuvers in Jupiter Space Pror to Ganymede Landing

3.8.3.2 Mission System Description

3.8.3.2.1 Mission Options

Not applicable.

3.8.3.2.2 Payload Descriptions

Lander System—The lander system is based on the Surveyor vehicle, with the solar cell arrays replaced with RTG's. Also some changes in insulation are required, resulting in a modest increase in weight. The orbital element is based on the planned Mariner Jupiter Orbiter (MJO) 1981 spacecraft, without the propulsion package (since the propulsion module defined here provides all ΔV). A mass summary for the lander and orbiter is presented in table 3.8-13. The configuration including propulsion package is illustrated in figure 3.8-9.

Table 3.8-13. Ganymede Mass Summary

Component	kg	lb
Lander inerts	310	684
Lander vernier propellants and gasses	100	220
Initial lander weight	410	904
Orbiter	740	1,631
Propulsion module inerts	400***	882
Propellant for Ganymede capture: ΔV^{*} = 4 080 M/S (13,382 fps)	2 200	4,850
Propellant for powered swingby: ΔV^{**} = 1 190 M/S (3,903 fps)	1 750	3,858
Total****	5 910	13,029

* With lander only, does not include vertical terminal vernier provided by lander.

** With lander and orbiter.

*** Power and communications through orbiter.

**** Excluding Earth departure and midcourse propulsion.

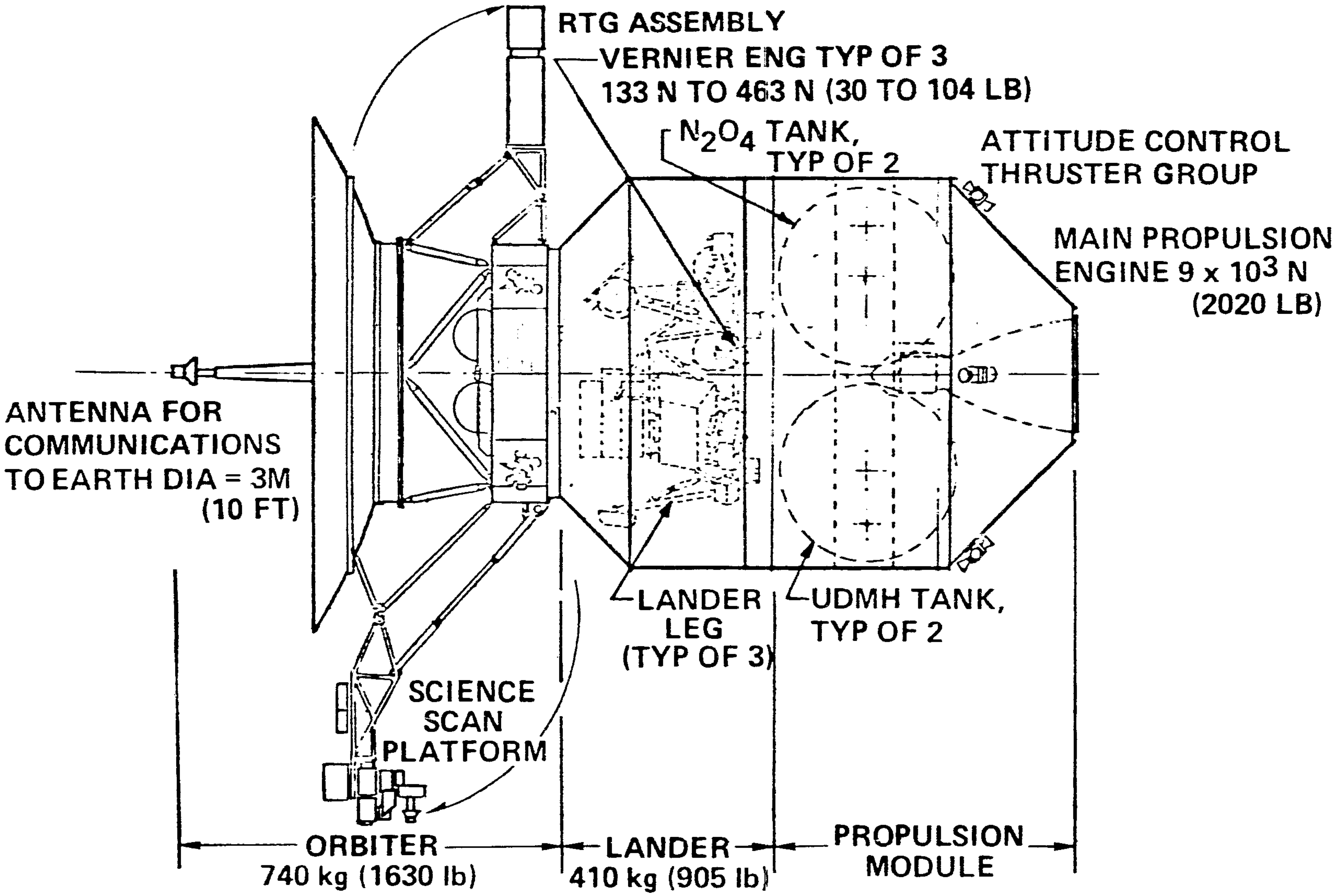


Figure 3.8-9 Ganymede Probe System Elements

Crew Rotation and Resupply—Not applicable.

Pickup Points and Transportation Constraints—The payload is designed to be launched integrated with a propulsion stage, and protected from Earth ascent aerodynamic loads.

3.8.3.2.3 Transfer and Storage

Not applicable.

3.8.3.2.4 Orbital Assembly, Maintenance and Modification

Not applicable.

3.8.3.3 Transportation Requirements

3.8.3.3.1 Payload Delivery Points

The Ganymede lander requires only injection to a Jupiter transfer. Successful execution of the mission plan depends on a slow transfer injection with C_3 about 80 km^2/sec^2 ($6.98 x 10^{20}$ ft^2/sec^2), requiring a departure delta V of about 6 500 m/sec (21,325 ft/sec) including gravity losses.

3.8.3.3.2 Payload Delivery Options

Single-stage LO_2/LH_2 and LO_2/MMH OTV's were examined.

3.8.3.3.3 Operational Constraints

Transfer opportunities to Jupiter occur about every 13 months and last approximately 20 days.

3.8.3.4 Mission/Transportation Modes and Operations

The Ganymede lander system at 5 910 kg (13,000 lb) requires an LO_2/LH_2 OTV of approximately 36 000 kg(79,000 lb) to deliver it to the required Jupiter transfer injection. If an LO_2/MMH OTV is used, the required mass is 56 700 kg (125,000 lb).

The Ganymede lander mission with either of the transportation options appear practical in all respects.

3.8.4 OVERALL AUTOMATED PLANETARY TRANSPORTATION COMPARISON

Required OTV masses are compared in Figure 3.8-10. Several of the options are seen to fall within the ranges of requirements determined for the manned program options. Required numbers of launches are summarized in Figure 3.8-11.

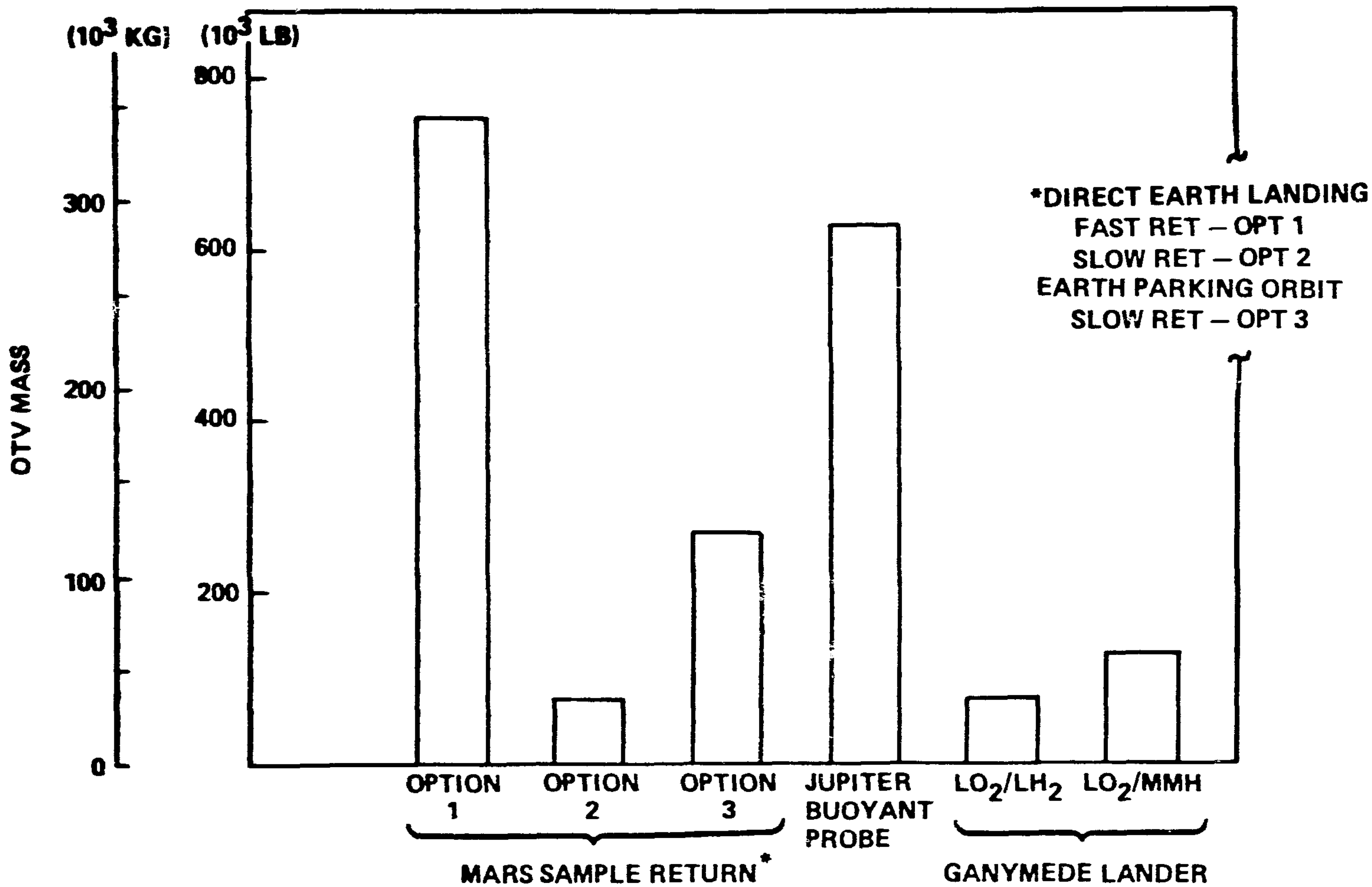


Figure 3.8-10. OTV Mass Comparison for Automated Planetary Missions

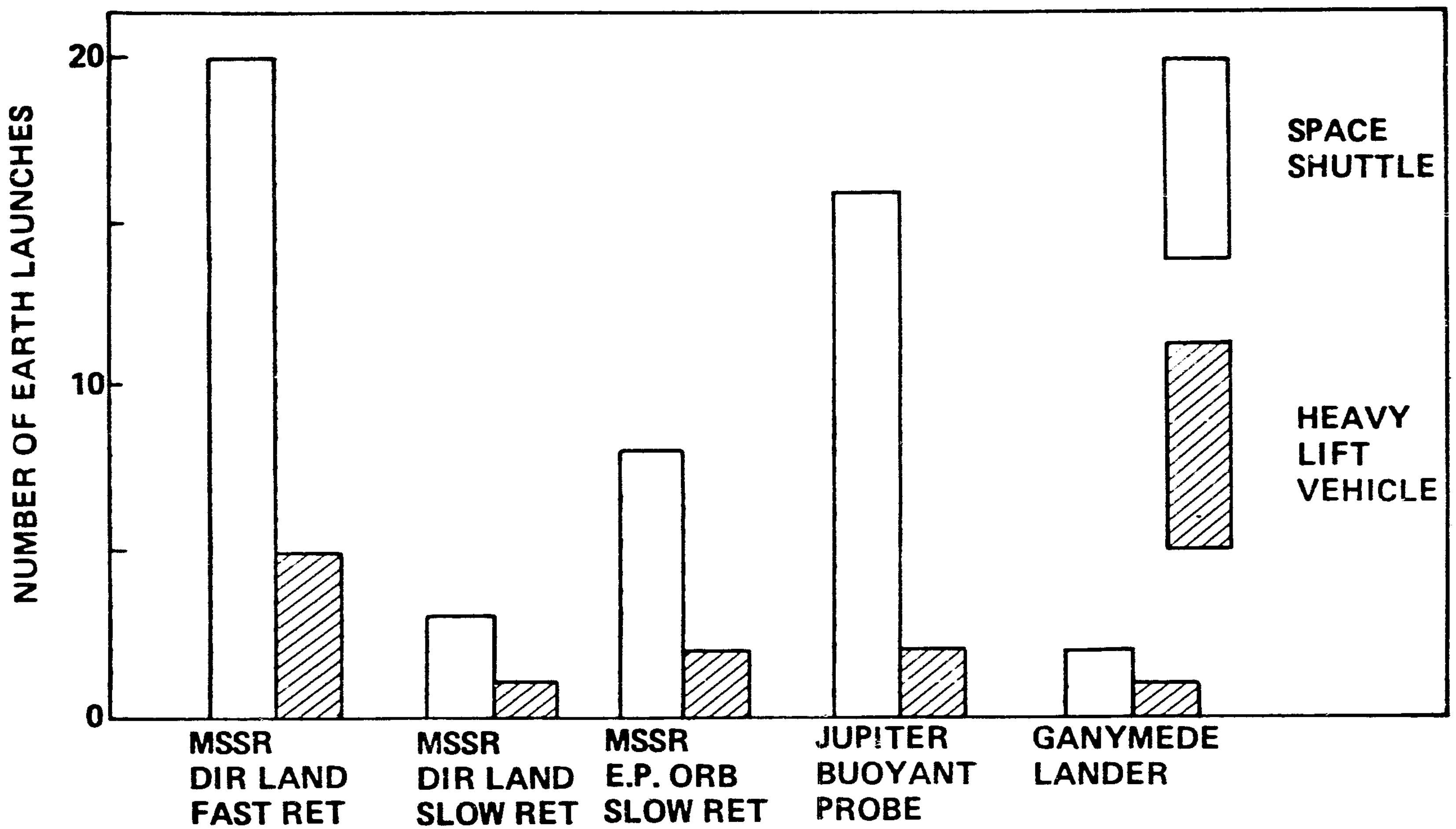


Figure 3.8-11. Earth Launch Requirements Summary Automated Planetary Missions

3.9 NUCLEAR WASTE DISPOSAL IN SPACE

Production of nuclear energy by fission results in production of highly radioactive wastes consisting of fission products and radioactive, mostly nonfissionable isotopes created by neutron capture. Nuclear waste material is initially millions of times more radioactive than the parent uranium. It decays, roughly exponentially, to a level below that of the parent in about 10 million years. There is no general agreement on the length of time required for decay to a "safe" level. Estimates of various authors range from "hundred of years" to "millions of years."

A number of methods for permanent (i.e., not requiring a continuing monitoring program) disposal have been studied. One of these is disposal in space.

The power produced per unit of waste product produced by nuclear energy is large, leading to an economic leverage potentially large enough to permit the cost of space disposal. Most studies of space disposal, in order to enhance the economics, have dealt with refined wastes (short-lived isotopes separated out and held in monitored storage until they decay to safe levels). A system to dispose of total waste would be desirable if feasible.

3.9.1 REFINED WASTE

3.9.1.1 Mission Summary

3.9.1.1.1 General Description

Nuclear waste materials as normally processed consist mostly of inert non-radioactive material. The radioactive components are principally fission products with half-lives of 30 years of less. These will decay to harmless levels in less than 1,000 years. Also present in much smaller quantities are *actinides* (named for the element actinium); they are the result of non-fission neutron capture by uranium and plutonium. Included are isotopes of neptunium, americium, californium, etc. These isotopes are comparatively well suited to space disposal: the mass to be delivered is relatively small, the isotopes are easy to shield, and importantly, these isotopes have very long half lives and are highly toxic. They present the principal challenge to disposal on Earth because they are hazardous for up to ten million years, longer than safety of Earth disposal methods can be predicted with any confidence. Separation of the actinides from the remainder of the waste material, packaging them suitably, and launching them to a space destination for disposal, comprises the baseline approach of refined waste disposal. Figure 3.9-1 illustrates the concept.

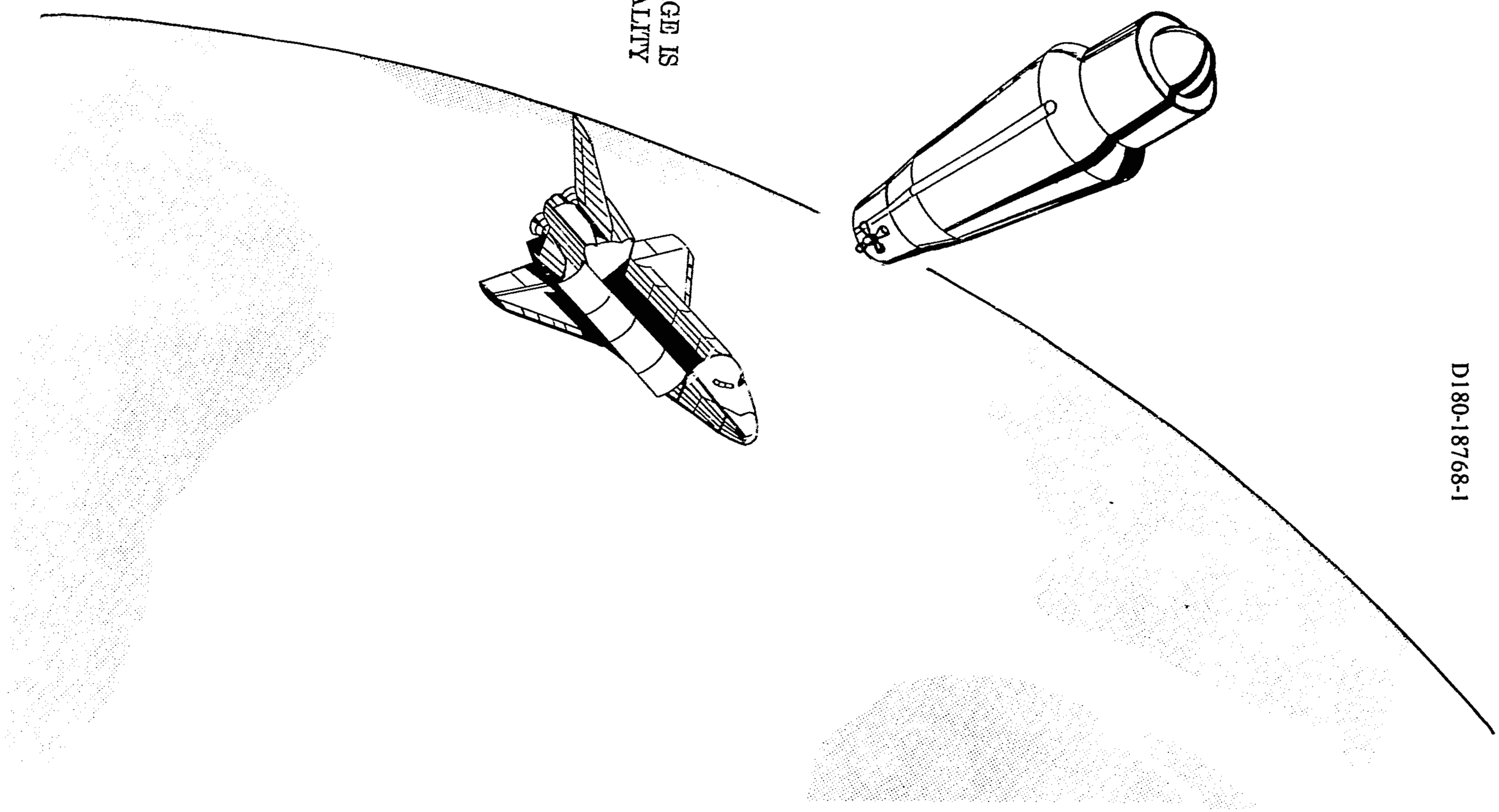


Figure 3.9-1: Orbital Deployment of Refined Waste Disposal Vehicle

3.9.1.1.2 Mission Assumptions and Constraints

Assumptions and constraints are presented in table 3.9-1.

Table 3.9-1. Nuclear Waste Disposal Mission Assumptions

Mission	Objectives	Mission Assumptions and Constraints
In-space Disposal	Effectively and safely remove from the Earth the total U.S. production of radioactive nuclear actinide wastes.	• Destination orbit to be heliocentric, and confined between 0.80 and 0.90 astronomical units. Alternative option direct solar system escape • Use waste production rate from nuclear power projections for 1985 (250,000 megawatts electric) to estimate traffic for 1995, i.e. 10-year delay for processing and short-term storage.

3.9.1.2 Mission Systems Description

3.9.1.2.1 Mission Options

3.9.1.2.1.1 Chemical Propulsion

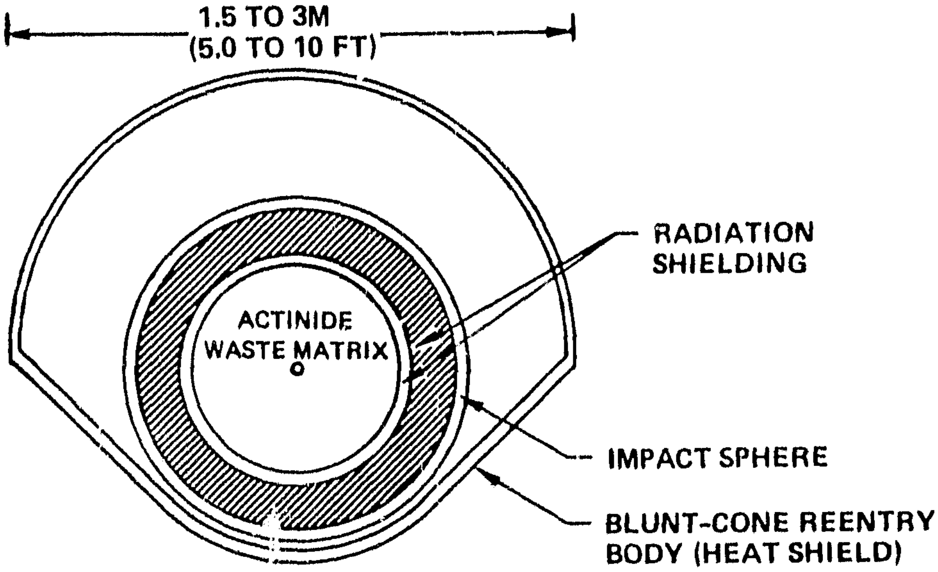
The use of ordinary chemical propulsion systems is the baseline approach, based on studies performed by the Lewis Research Center.

3.9.1.2.1.2 Electric Propulsion Utilizing Decay Heat

The Marshall Space Flight Center has conducted an inhouse study in cooperation with elements of the Energy Research and Development Agency (ERDA) to define a candidate system for utilizing the decay heat of actinide wastes. The decay heat would operate a thermionic conversion system to generate electric power to drive an electric propulsion system capable of propelling the wastes from a low Earth orbit to solar system escape. This option was not analyzed by this study and is not discussed further. A brief analysis of the use of the decay heat from total solidified waste was performed. It was found that the energy content of total waste is marginal. The energy content of (refined) actinide wastes is more than ten times greater per unit mass than that of total wastes and is more than sufficient.

3.9.1.2.2 Payload Descriptions

The concept shown in figure 3.9-2 was developed by the Lewis Research Center through inhouse study. Waste is refined to separate out the long half-life actinides. Only these are launched to space disposal. The disposal package incorporates entry and impact protection plus sufficient shielding for safe handling and approach. It is therefore potentially compatible with launch by an unmanned, non-intact-abort vehicle. The package size shown is representative; package size can be selected within reasonable limits to match the transportation system.



WEIGHT BREAKDOWN OF TYPICAL
NUCLEAR WASTE PACKAGE
[SOLAR SYSTEM ESCAPE FOR ACTINIDES.]

COMPONENT	MASS	
	LBS	KG
ACTINIDE WASTE	440	200
MATRIX CONTAINING WASTE	1375	625
GAMMA SHIELD	2618	1190
NEUTRON SHIELD	396	180
IMPACT SPHERE	1408	640
REENTRY BODY (HEAT SHIELD)	902	410
TOTAL	7139	3245

*Figure 3.9-2. Refined and Shielded Waste Concept**

*LEWIS RESEARCH CENTER

Crew Transfer and Resupply Not applicable.

Mass Summary Not applicable. Representative payload mass data were given in figure 3.9-2.

Pickup Points and Transportation Constraints The payloads are designed to be launched to Earth orbit integrated with the propulsion systems intended to launch the payloads to their destination.

3.9.1.2.3 Transfer and Storage

Not applicable.

3.9.1.2.4 Orbital Assembly, Maintenance, and Modification

Not applicable.

3.9.1.3 Transportation Requirements

3.9.1.3.1 Payload Delivery Points

Many alternative destination options (i.e., where to send the waste) have been proposed and discussed. The destination options selected as representative are a solar orbit at 0.9 AU (between the Earth and Venus) and direct solar system escape. These span the range of destination energies. High Earth orbits have been proposed; they require about the same energy as the 0.9 AU solar orbit. Solar systems escape by Jupiter flyby requires less energy than direct escape and has been proposed; in fact, the spacecraft Pioneer 10 and Pioneer 11 are on such trajectories. However, the Jupiter launch window is open only during a 20- to 30-day period every 400 days. The traffic level required to utilize this window while it is open is very high, making such a consideration impractical.

3.9.1.3.2 Payload Delivery Options

Not applicable.

3.9.1.3.3 Operational Requirements and Constraints

Nominal operational requirements are stated in table 3.9-2.

Table 3.9-2. Operational Requirements

Mass requirements	
Representative waste package — kg (lb)	3,221 (7,150)
Package size — m (ft)	2 x 1.8 (6.6 x 6)
Rate requirements for 250,000 megawatts power	
Packages/yr	50
Mass/yr — kg (lb) (packages only)	163,636 (360,000)

Special requirements exist due to the hazardous nature of the material transported:

- Public exposure to the material must be prevented. This is accomplished in the baseline system by the packaging system.
- The flight crew must be protected from radiation exposure by a suitable combination of shielding plus operational procedure.
- Waste packages must be under strict control at all times. Unmanned launch by a single vehicle delivering a package to orbit followed directly by injection to the destination is permissible. Operations through injection to the destination must be continuously monitored and controlled to preclude inadvertent re-entry of the waste packages into the atmosphere.

3.9.1.4 Mission/Transportation Modes and Operations

3.9.1.4.1 Transportation Options

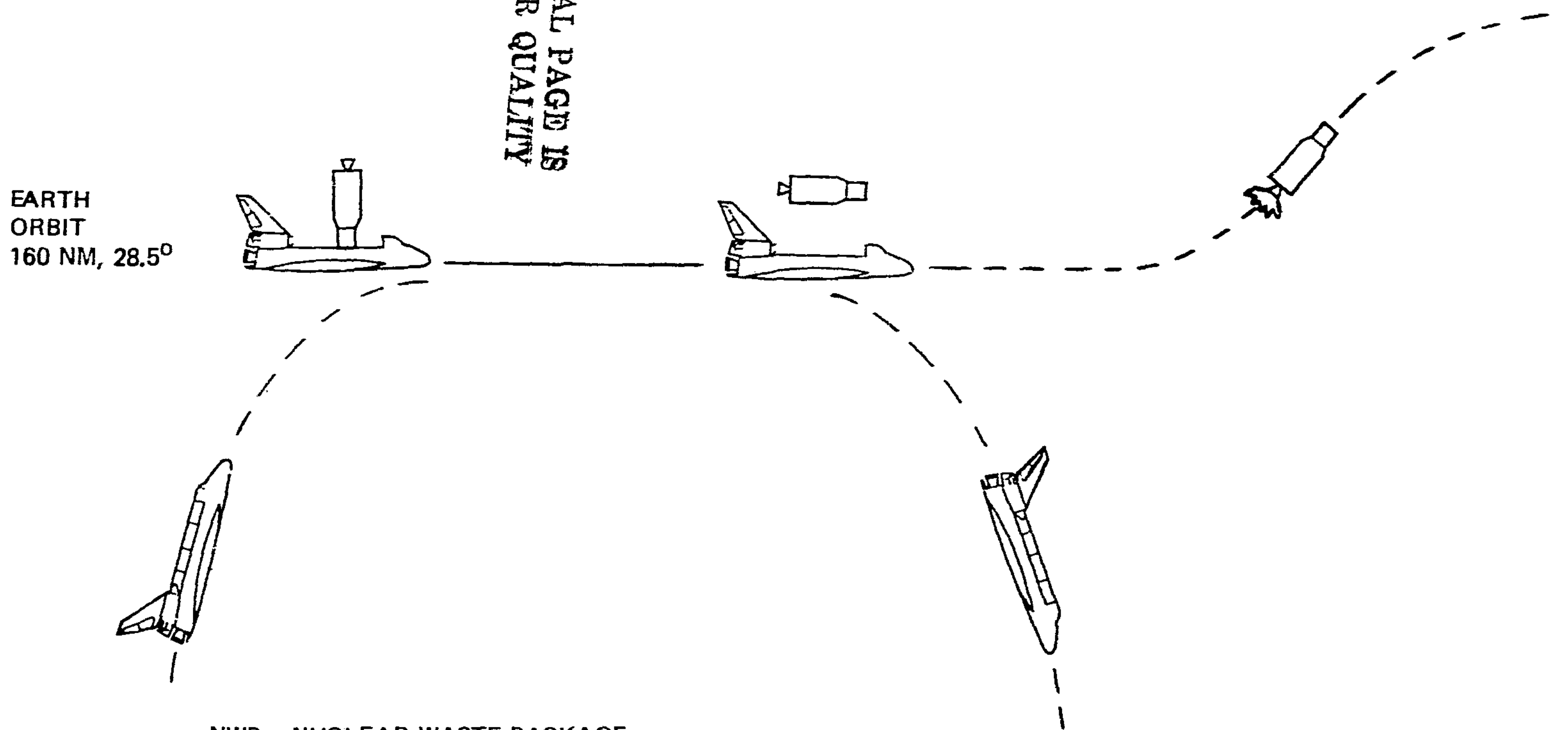
Alternatives were not analyzed.

3.9.1.4.2 Representative Transportation Mode and System

3.9.1.4.2.1 Transportation Sequence

Figure 3.9-3 shows the transportation sequence for the Lewis Research Center concept for launch to 0.9 AU. Figure 3.9-4, repeated from the Lewis Research Center study, shows the transportation sequence for solar system escape (SSE). The 0.9 AU mission takes one shuttle flight per waste package; the SSE mission, two. The full-capacity tug for the 0.9 AU mission can be reused by using a solid rocket motor kick stage for the solar orbit circularization burn of 1,700 m/sec (5,580 ft/sec). The tug injects the SRM plus payload to a C_3 of 2.5 km^2/sec^2 (6.73 x 10^7 ft^2/sec^2), performs a retro maneuver of approximately 500 m/sec (1,640 ft/sec) to become recaptured in Earth's gravity potential, and upon return to low orbit perigee performs a braking maneuver of approximately 3,300 m/sec (10,800 ft/sec) to circularize in low orbit. If the tug is expended, it burns in a single burn to a C_3 of 2.5 km^2/sec^2 (6.73 x 10^7 ft^2/sec^2) and separates from the SRM/waste payload.

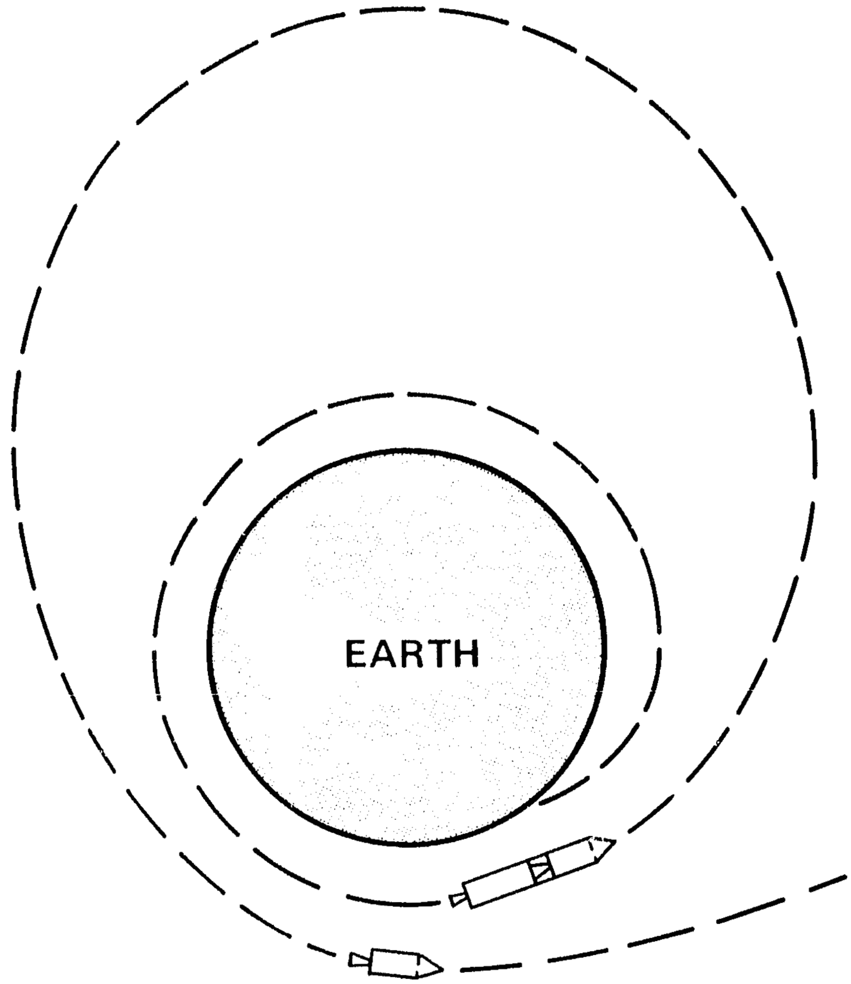
Tables 3.9-3 and 3.9-4 present sequence summary data for the 0.9 AU (reusable tug) and SSE modes.



NWP = NUCLEAR WASTE PACKAGE

• DELIVER OTV/NWP TO ORBIT WITH SS	• CHECKOUT • DEPLOY OTV/NWP	• INJECT NWP INTO 0.9 AU SOLAR ORBIT OR TO SOLAR SYSTEM ESCAPE USING OTV • REPEAT APPROX. ONCE PER WEEK

Figure 3.9-3. Nuclear Waste Disposal–Lewis Research Center Concept



SOLAR SYSTEM ESCAPE. VELOCITY INCREMENT, ΔV = 9.16 KM/SEC (30,000 FT/SEC); TWO SHUTTLE LAUNCHES TO 370 KM (200 N MI) ORBIT (ONE SHUTTLE CARRIES PAYLOAD AND EXPENDABLE TUG, THE OTHER CARRIES REUSABLE TUG); TWO BURNS AT PERIGEE; TIME BETWEEN BURNS, ~ 8 HR

Figure 3.9-4. SSE Mission Sequence

Table 3.9-3. NWD Mass History Using Tug (0.9 AU)

Event	Time (hr)	Delta V		Mass Remaining	
		M/Sec	Ft/Sec	Kg	Lb
Initial mass	0	–	–	27 660	60,980
Inject with tug	0.2	3,415	11,200	12 890	28,420
Separate tug	1	–	–	6 490	14,310
Circularize at 0.9 AU	4350	1,700	5,580	3 530	7,780
Tug mass at separation	1	–	–	6 400	14,[illegible]
Retro burn	1.2	–	–	5 700	12,565
Return orbit insertion	10	3,300	10,825	2 725	6,005

Table 3.9-4. NWD Mass History Using Common-Stage Mode for Solar System Escape

Event	Time (hr)	Delta V		Mass Remaining	
		M/Sec	Ft/Sec	Kg	Lb
Initial mass	0			57 370	126,472
Boost	0.5	2,240	7,350	28 624	63,105
Drop Stg 1				24 240	53,440
Inject with Stg 2	8	6,920	22,700	5 970	13,161
Drop Stg 2				3 245	7,154
Stg 1 + fuel				4 385	9,667
Stg 1 retro into LEO	8	2,140	7,020	2 725	6,005

3.9.1.4.2.2 Transportation Sizing

Both sequences described above match well with the nominal full-capability tug (FCT). If the tug is expended on the 0.9 AU mission, larger packages can be handled, resulting in a need for approximately 30 packages per year rather than 50.

3.9.1.4.2.3 Operational Factors

The nuclear waste mission requires protection of the public and the biosphere in general from accidental release of the waste. A thorough analysis of nuclear safety was beyond the scope of this study. The modes selected were chosen with the belief that nuclear safety requirements could be accommodated, but this was not demonstrated.

The 0.9 AU destination requires a second burn six months after Earth departure. This mission therefore requires tracking of dozens of packages at once. The SSE mission requires no tracking once the energy for SSE is established.

3.9.1.4.2.4 Earth Launch Requirements Summary

The 0.9 AU mission requires 30 shuttle launches annually with tug expended or 50 with tug reused. The SSE mission requires 100 shuttle launches annually. If a HLLV is used, about one-fourth as many launches are needed. All these values are based on the assumption of 250,000 megawatts of generation capability.

3.9.1.4.3 Transportation Options Comparison and Evaluation

Comparisons were not developed, except as discussed above. All the options described above appear practical and all can be done with a very modest surcharge on electric power, less than $0.001 (1 mill) per kwh.

3.9.2 TOTAL WASTE

A brief analysis of total waste disposal options was carried out and is reported in the appendix to this report.

3.10 SATELLITE ENERGY SYSTEMS

Two classes of satellite systems related to terrestrial utilization of electric power have been proposed:

- The first is on-orbit power generation with power transmission by microwave beam to an Earth receiving station. Solar energy is converted to electricity by the satellite employing either of two conversion system options that have been analyzed, solar photovoltaic cells and thermal concentrator heat engines. Figure 3.10-1 shows the system concept.

- The second is on-orbit reflectors (power relays), for microwave beams generated by Earth-based power and received by Earth-located stations near the power distribution point. Note that this is a concept for power transmission, not generation. Figure 3.10-2 shows the power relay concept.

These systems would normally be located in geostationary (geosynchronous) orbit, although other locations have been proposed. Analyses in this study have been confined to the geosynchronous location.

The systems depend upon large-scale highly efficient transmission of energy by microwaves. This is theoretically possible and preliminary feasibility experiments have been encouraging. Analysis of microwave power transmission is beyond the scope of this study.

3.10.1 POWER GENERATION SYSTEMS

3.10.1.1 Mission Summary

3.10.1.1.1 General Description

Large satellites for collection and conversion of solar energy are to be placed in a geosynchronous orbit. Each satellite will be capable of delivering from 2 000 to 20 000 megawatts of electric power to commercial networks through a ground receiving station. Most studies to date have considered either 5 000 or 10 000 megawatt satellites. Microwave transmission link efficiencies on the order of 70 percent have been predicted; assuming that these are realized, the electric power generation capability of the satellite will be about 1.4 times the ground station output.

Figure 3.10-3 shows the solar photovoltaic satellite concept pioneered by Glaser, et al; figure 3.10-4 illustrates a solar heat engine concept developed in Boeing IR&D studies.

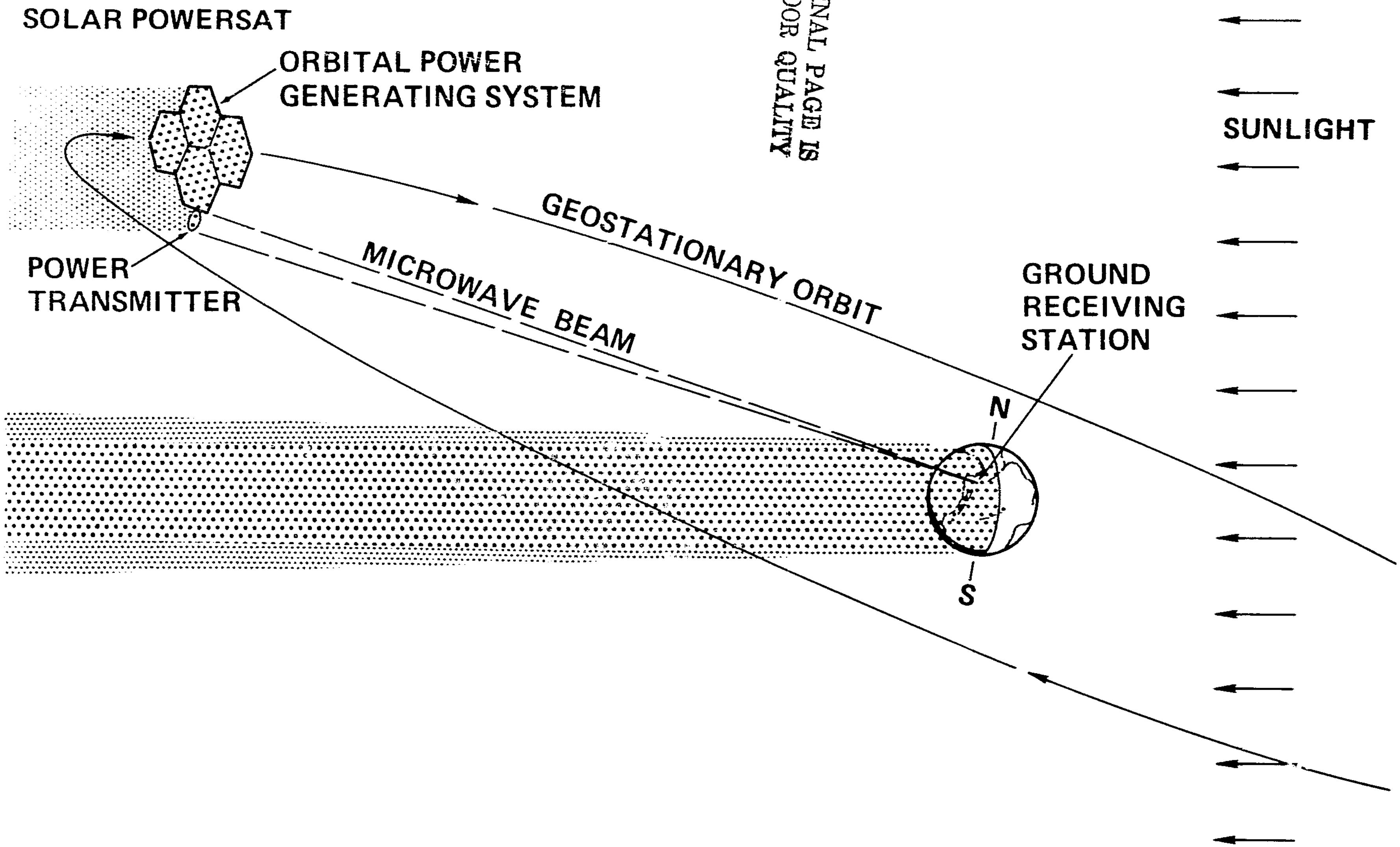


Figure 3.10.1. Solar Power Satellite Concept

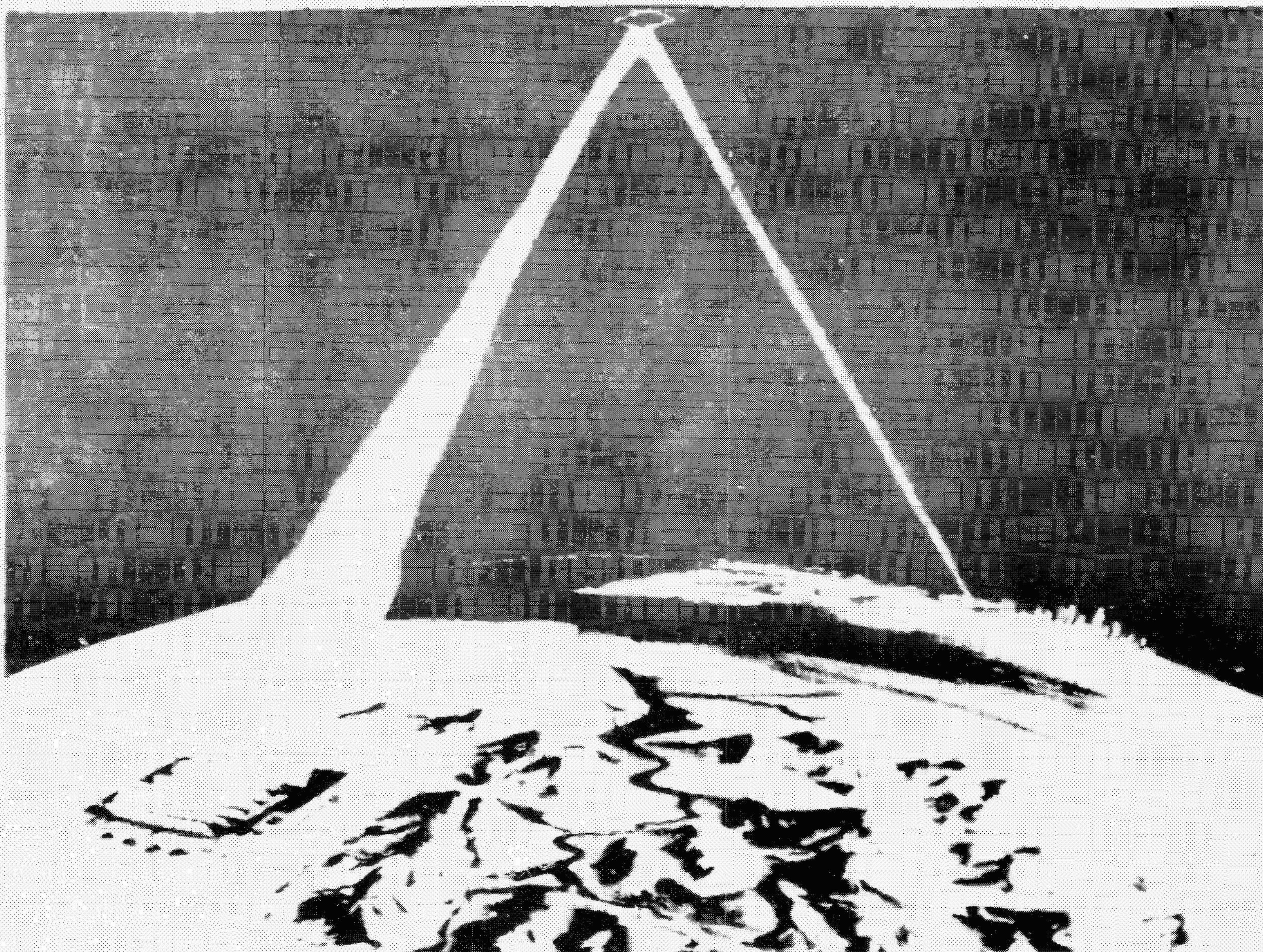


Figure 3.10-2. Microwave Power Relay Concept

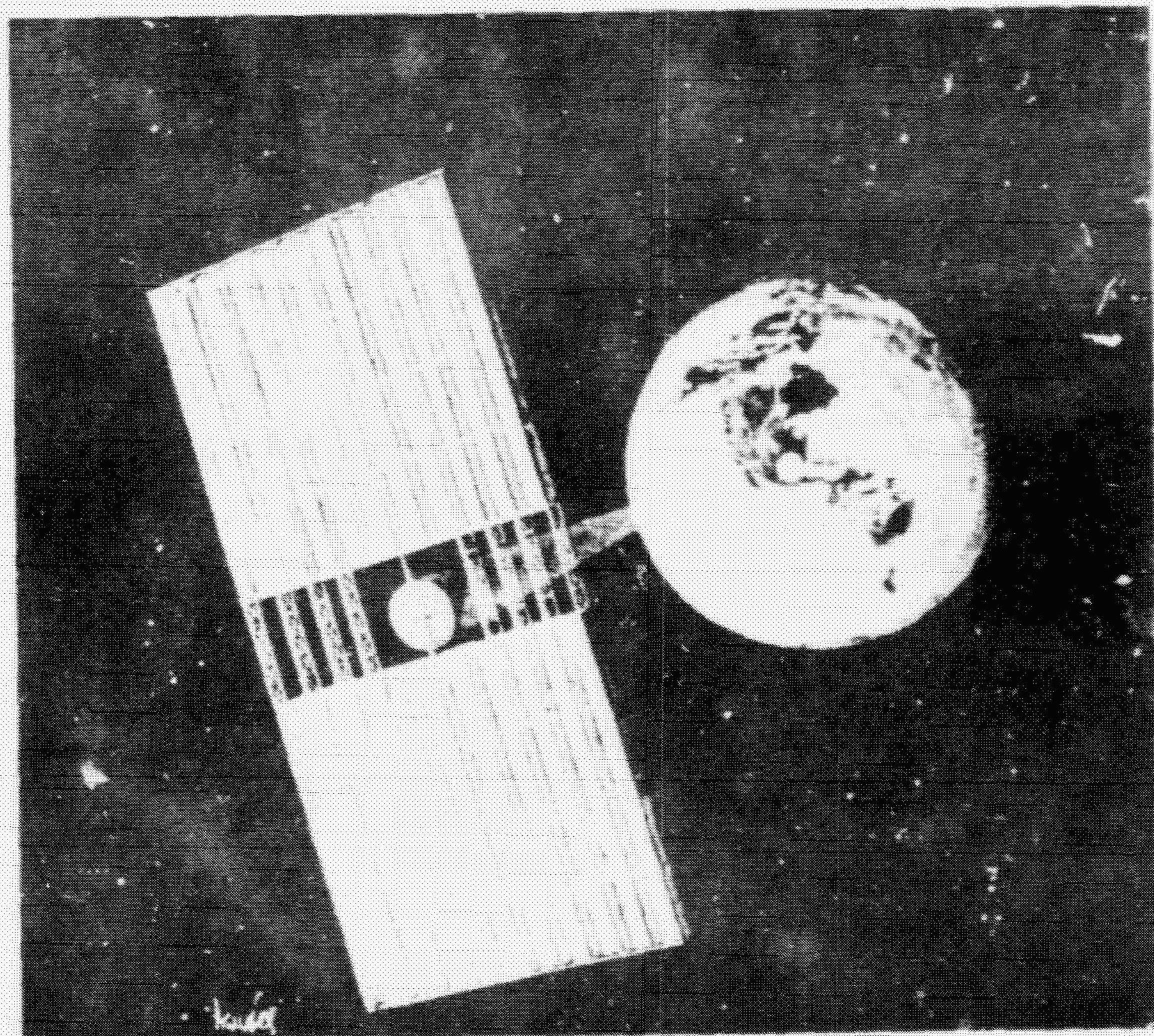


Figure 3-10-3. Photovoltaic Power Satellite

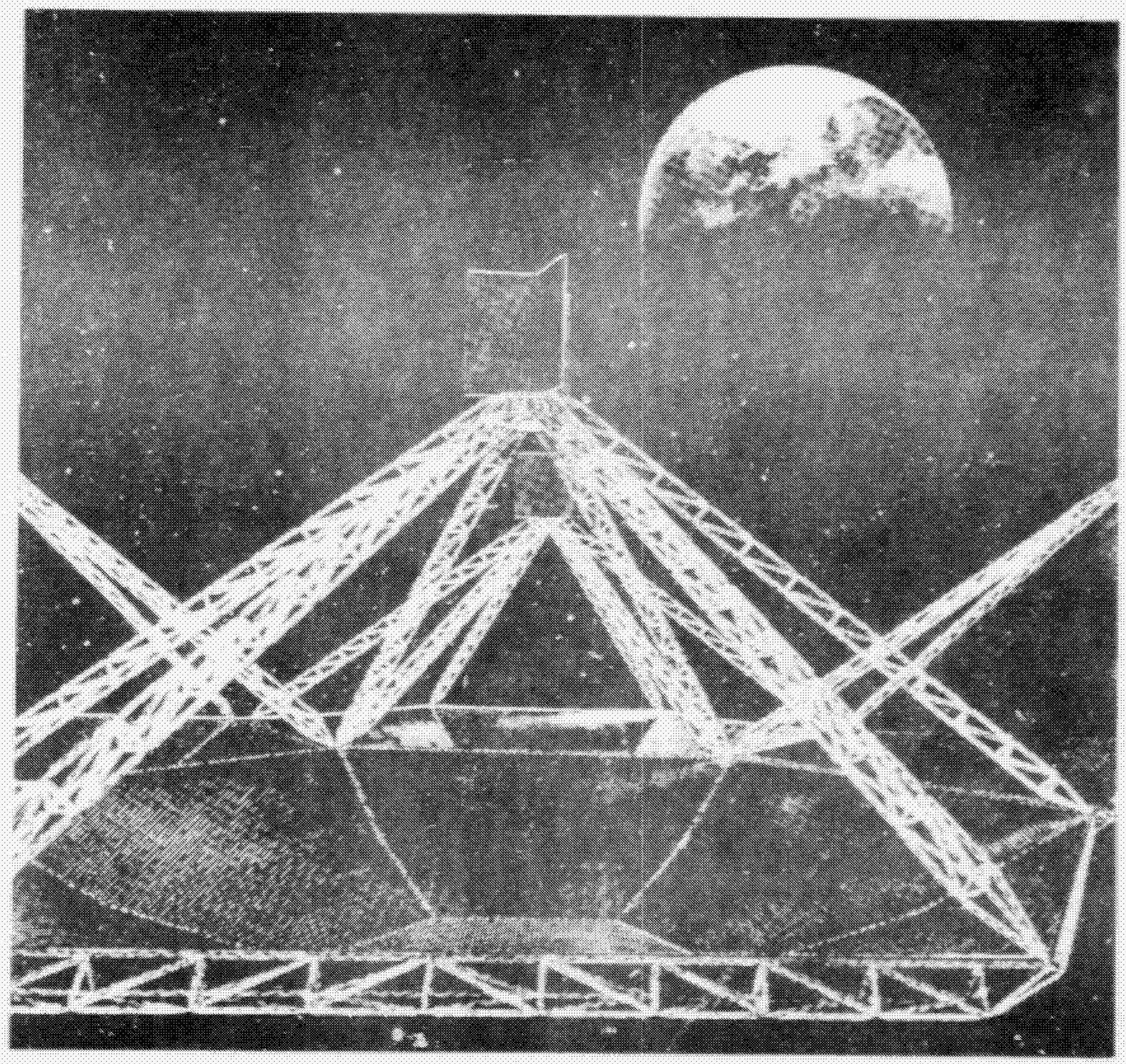


Figure 3.10-4. Thermal Engine Power Satellite

3.10.1.1.2 Mission Assumptions and Constraints

It is generally assumed that power satellites must be capable of producing electric power at commercially acceptable costs. A representative figure is $0.025 per kwh in 1975 dollars. This in turn places cost related requirements on the satellites and their transportation systems.

Because of the commercial nature of the systems described, it is assumed that power satellites will be placed on orbit at a rate equal to the demand for new plant capacity in this form. Various projections have been made, ranging from less than one to several 10 000 megawatt satellites per year.

3.10.1.2 Mission System Description

The systems described are fully operational satellites. The resulting program is quite large in scope by comparison to the others reported in previous pages. A demonstration program, aimed at developing and demonstrating all the key capabilities and technologies needed for a power satellite program, would in itself result in a substantial transportation requirement and one that would of course arise before that of the operational program. Demonstration programs have not yet been defined, but some activity is currently taking place to do so. As representative demonstration programs become adequately defined, they will be included in subsequent phases of the study. An operational satellite energy system includes many elements besides the satellite itself. Table 3.10-1 summarizes system elements.

3.10.1.2.1 Mission Options

3.10.1.2.1.1 Photovoltaic Generation

Figure 3.10-5 shows dimensions and general layout of the photovoltaic satellite defined in NASA CR-2357. Figure 3.10-6 shows a typical detail. Aluminized plastic film reflectors are used to concentrate solar energy on the cells to reduce cell inventory. A concentration ratio of 2 is used; this is about as much as can be employed without active cooling of the cells.

3.10.1.2.1.2 Thermal Engine Generation

Figure 3.10-7 illustrates the thermal engine generation concept. Each generation module employs a concentrating reflector composed of 10,000 or more flat stretched aluminized plastic film reflector facets. Each facet is individually controlled to direct its reflected sunlight into a cavity absorber. This approach allows achievement of the high concentration ratios, 2,000 or more, needed for an efficient thermal engine while attaining the relatively low mass resulting from use of plastic film.

Table 3.10-1. Satellite Energy System Description

Work Breakdown Structure Level	Element	Function	Description	Cost Leverages
1	Satellite Energy System	• Collect & convert energy in space for commercial use on Earth.	Satellite solar energy collectors, converters in geosynchronous orbit; microwave transmission to Earth.	Total system investment cost must be minimized to minimize cost of power sold to users.
2	Satellite System	• Collect energy, convert to RF, and radiate to Earth.	• Sun-oriented satellites large enough to produce 5000 MW or more electric power at Earth receiving station.	• Cost of satellite is a major share of system cost. Mass of satellite drives transportation cost.
3	Generation System	• Convert solar energy to electric energy.	• a) Solar photovoltaic, or b) solar concentration heat engine system.	• Generation system cost and mass are the principal variables.
3	Microwave Power & Transmission	• Convert electric energy to RF (Microwave) form & radiate to Earth.	• Large phased array antenna with amplitron microwave generation; active phase control.	• Beam forming and RF generation efficiency impact size of power generation system.
3	Control & Support Systems	• Control satellite & provide support functions (Data, attitude,control, etc.)	• Control equipment for power system and attitude control, etc., including habitable control station.	• Not a major cost contributor.
2	Transport & Space Support Systems	• Transport spaceborne elements to destinations & support satellite operations.	• Transportation and support systems as described below.	• Cost of transportation is a large share of system cost.
3	LEO Freighter	• Transport major system elements to low Earth orbit.	• Reusable Earth-to-orbit carrier for low density payloads of 100,000 kg or more.	• Must achieve low cost, on the order of $20/lb, to make system feasible.
3	LEO Logistics Vehicle	• Transport crews & light cargo to and from low Earth orbit.	• Reusable Earth-to-orbit carrier for light payloads & crews. Horizontal landing. Maybe partly common with LEO freighter.	• Very low cost per flight needed.
3	GSO Freighter	• Transport major system elements to geosynchronous orbit–includes self-power as applicable.	• Reusable orbit-to-orbit transport system nominally electric propulsion.	• Propellant required directly impacts total transportation cost; trip time causes delay cost.
3	GSO Logistics	• Transport crews & light cargo to & from geosynchronous orbit.	• Reusable orbit-to-orbit system for fast trips, light payloads. Nominally chemical propulsion.	• Needed to minimize outage time due to failures.
3	LEO Support	• Provide crew housing & other operational support in low Earth orbit.	• Combined payload handling, crew habitability, and satellite module assembly fixture station.	• Front-end cost item but required to support orbital crews.
3	GSO Support	• Ditto in geosynchronous orbit.	• Crew habitability and operations support station.	• Same
2	Ground Systems	• Provide all ground-based operational & support functions.	• Ground facilities for support of satellite assembly & operations and receiving of power.	• Influence system efficiency and are significant part of overall cost.
3	Power Receiver & Distribution Facility	• Receive microwave power beam, convert to dc & then to grid–compatible ac	• Rectifying microwave antennas ("rectennas") and dc/ac conversion equipment.	• Receiving efficiency impacts required satellite power level & size.
3	Manufacturing & Assembly Facility	• Manufacture & subassemble powersat elements as required.	• Manufacturing facility at launch site.	• Large part of satellite cost incurred in this facility.
3	Launch & Recovery Facility	• Conduct all flight system launch & recovery operations.	• Launch, recovery, & flight control centers for all transportation system elements.	• Must contribute to efficient system operation.
3	Logistics & Support Facilities & Operations	• Manufacture propellant, & generally provide logistic & operational support.	• a) propellant plants & storage; b) transportation vehicle refurbish & service facility; c) central control; d) other functions.	• Major share of transportation cost incurred in these facilities.

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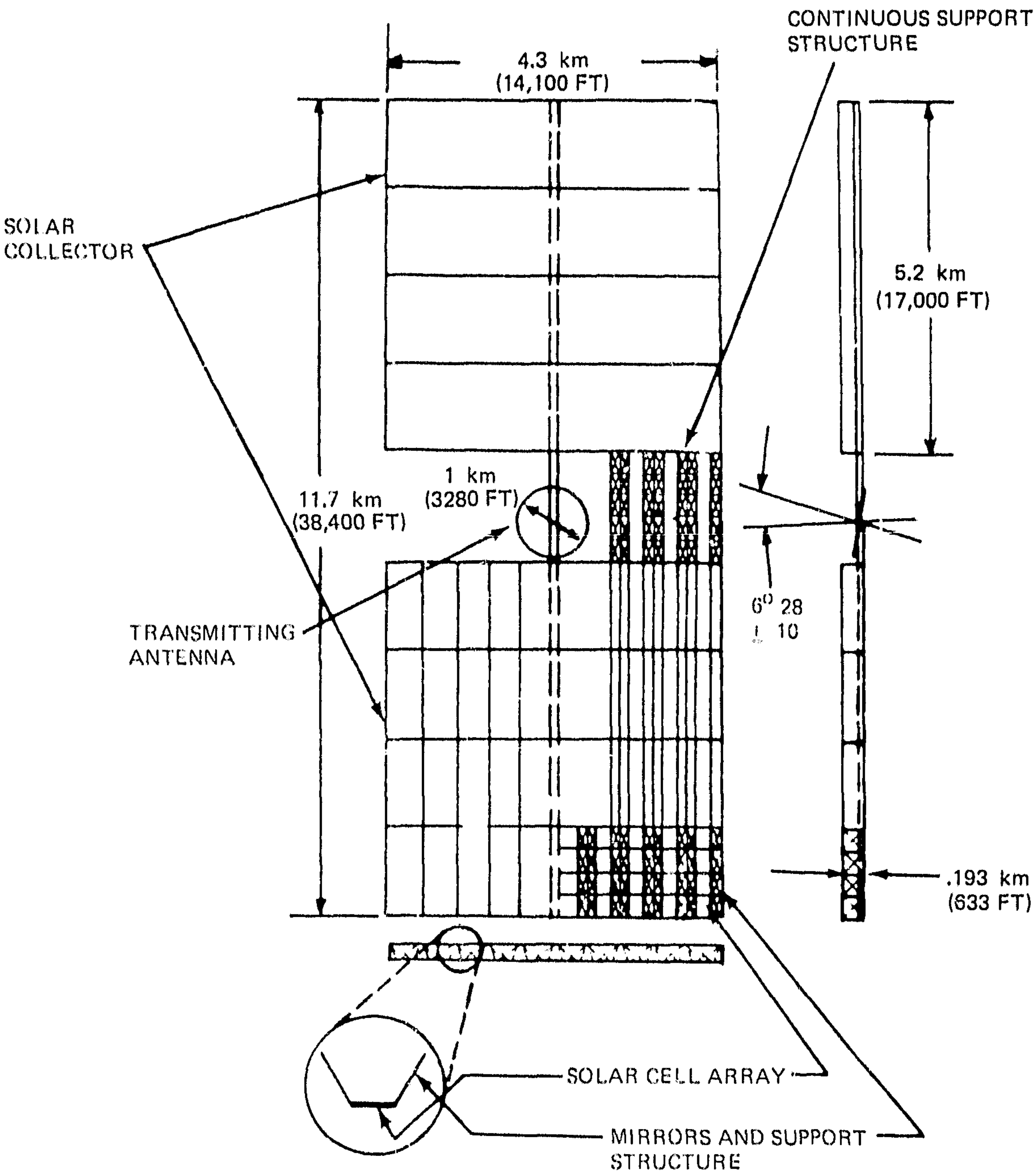

Figure 3.10-5. Photovoltaic Powersat

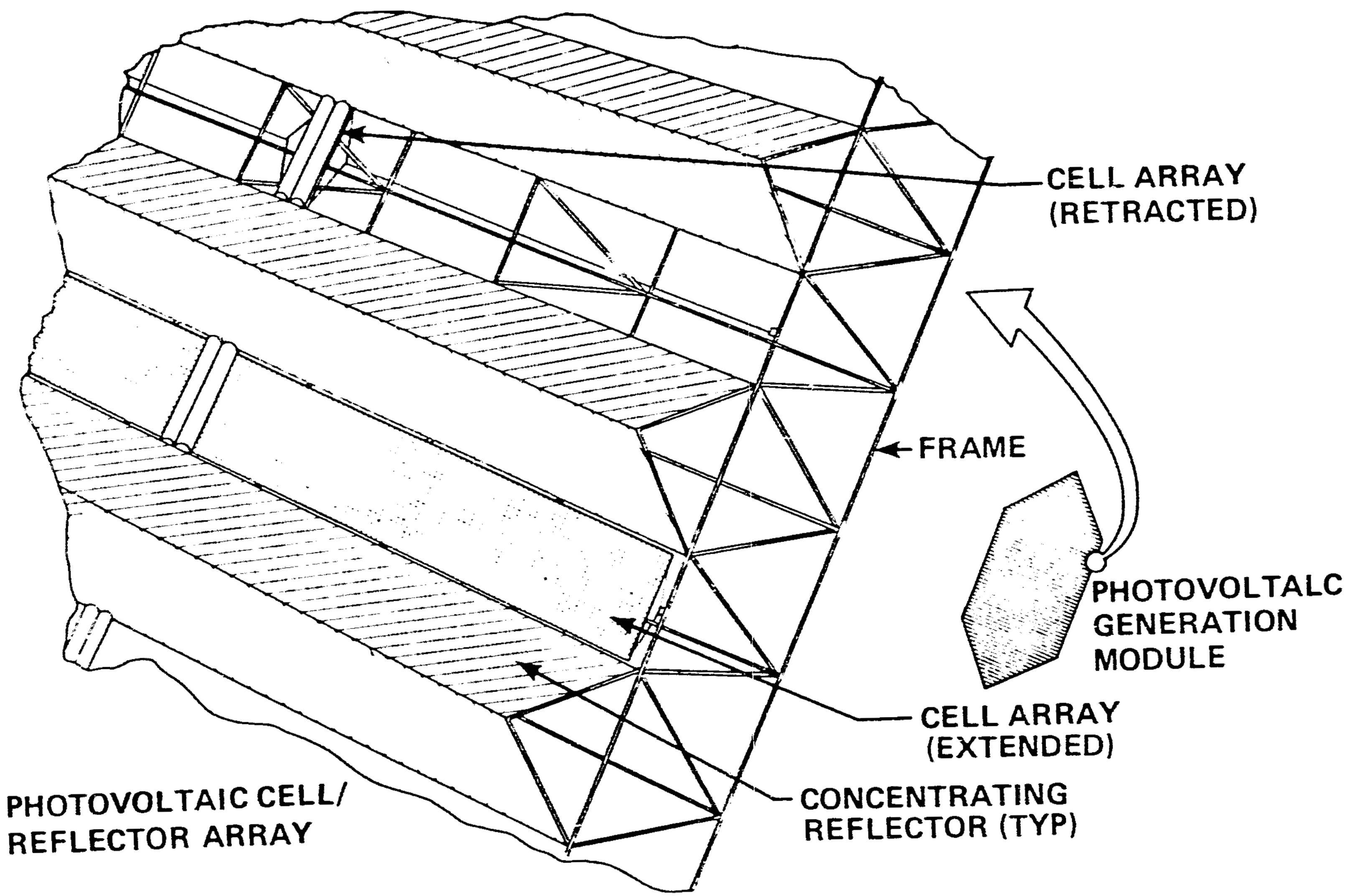

Figure 3.10.6. Photovoltaic Panel and Concentrator Arrangement

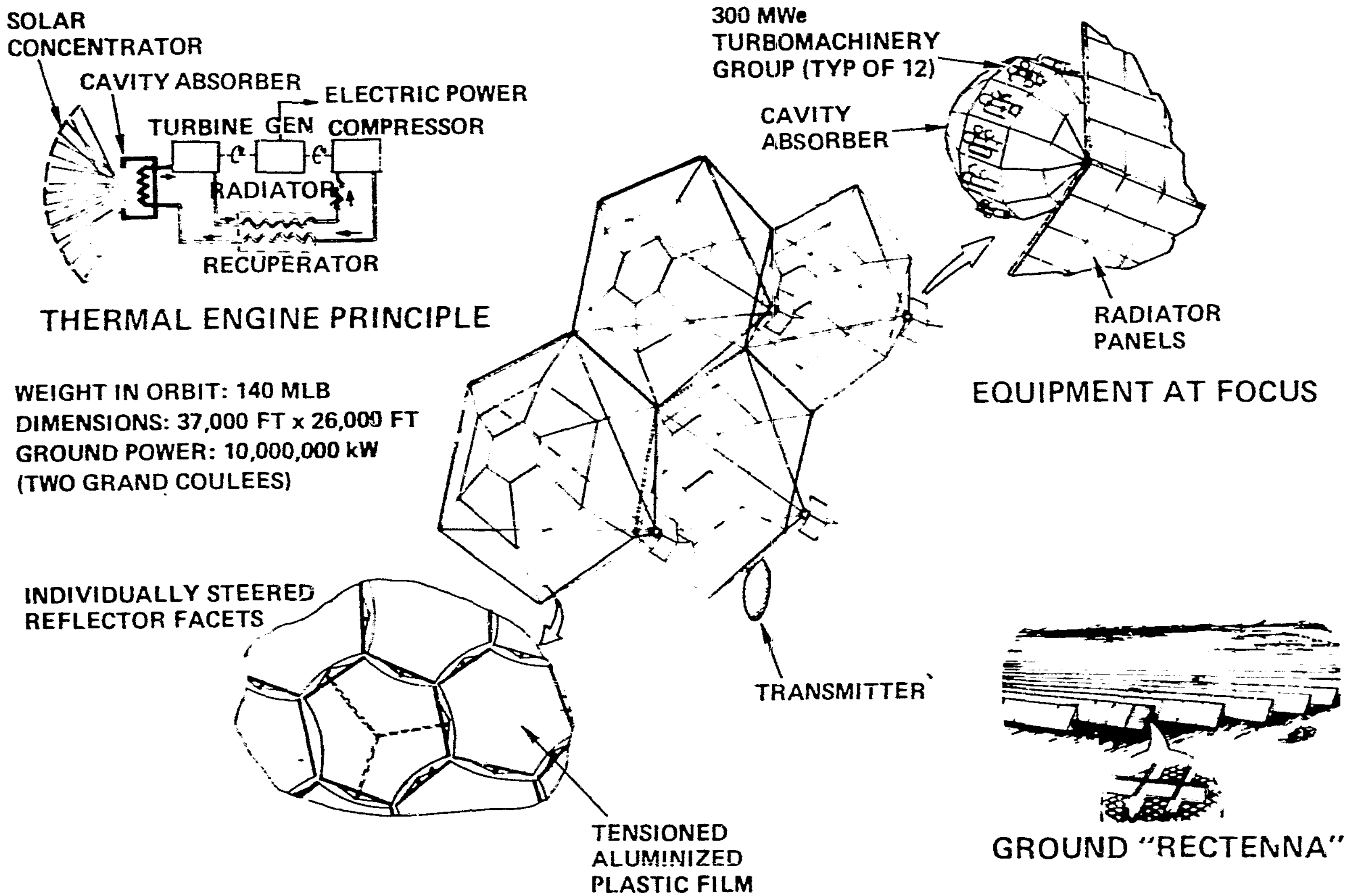

Figure 3.10-7. Thermal Engine Powersat Concept

The heat engine itself uses a closed cycle inert gas Brayton turboalternator. Unusable heat is rejected from the cycle by a large space radiator.

3.10.1.2.2 Payload Descriptions

3.10.1.2.2.1 Satellites

Various mass estimates for power satellites have been published. NASA CR-2357 reports 11 340 metric tons (25×10^6 lb) for an advanced technology photovoltaic system producing 5 GW (5000 MW) of usable power on the ground. Published Boeing IR&D studies have reported 34 000 metric tons (75×10^6 lb) for an advanced technology (high cycle temperatures) solar/thermal engine system producing 10 GW on the ground. Recent Boeing IR&D studies have developed estimates based on less advanced technology leading to greater masses (as would be expected). The range of available estimates is from approximately 2.3 to approximately 6.5 kg/kw (5 lb/kw to 14 lb/kw), (satellite mass/usable Earth power).

3.10.1.2.2.2 Other

Payloads in addition to the powersats themselves will include assembly equipment, transportation system(s) for transfer to geosynchronous orbit, crews, crew quarters, and consumables. These requirements are poorly defined at present; for the purposes of this study, it will be assumed that transportation systems capable of delivering the basic powersats will be able to meet this additional requirement.

Payload recovery requirements are small compared to delivery requirements; these requirements are well within the capability of the space shuttle.

3.10.1.2.2.3 Crew Rotation and Resupply

These requirements have not been well characterized by studies to date. It has been roughly estimated that an assembly support facility in low orbit will house a crew of 60 or more, and that a small crew in geosynchronous orbit will also be required. A composite requirement roughly the sum of space base (para. 3.1.3), geosynchronous manned station (para. 3.2.1), and geosynchronous satellite maintenance sortie (para. 3.2.2) is indicated.

3.10.1.2.2.4 Mass Summary

Insufficient data exist to put together a comprehensive mass summary. The power satellites themselves are the principal requirement.

3.10.1.2.2.6 Pickup Points and Transportation Constraints

The power satellites will be assembled in orbit from subassemblies and piece parts. These will be packaged on pallets for delivery to low orbit in a protected payload bay.

Major assemblies, modules, or complete satellites will be delivered from the low orbit assembly support facility to geosynchronous orbit. Pickup points have not been defined and will depend on the nature of the orbit transfer system selected.

3.10.1.2.3 Transfer and Storage

Payloads will be stored at the low orbit assembly facility to await use in the assembly process.

3.10.1.2.4 Orbital Assembly, Maintenance, and Modification

A major orbital assembly operation is indicated, requiring a dedicated facility. One concept of such a facility is shown in figure 3.10-8, showing a thermal engine power satellite module being constructed. The assembly support facility includes attachment to the satellite module, crew quarters, docking provisions, payload and propellant storage, power, and other features. A space shuttle can be seen in the illustration, docked to the facility. Also shown are assembly vehicles, crew work vehicles, etc.

3.10.1.3 Transportation Requirements

3.10.1.3.1 Payload Delivery Points

Satellites are delivered to low orbit in parts, assembled there, and delivered to geosynchronous orbit where further assembly may be required. Crew and support systems must be delivered to low orbit, geosynchronous orbit, and Earth return.

3.10.1.3.2 Payload Delivery Options

Packaging for delivery to the low orbit will depend upon the launch system selected.

Three assembly/transportation modes are available:

1. Low-orbit assembly of the total satellite with integral transfer to geostationary orbit
2. Low-orbit assembly of modules and modular transfer to geostationary orbit
3. All assembly in geostationary orbit

Low altitude assembly has the following advantages:

- Assembly crews and equipment (i.e., "orbital tooling") need not be carried to geostationary orbit thus minimizing the total energy required per plant produced.
- The system can be checked out before transfer so that corrections, replacement, etc. are more easily accomplished.

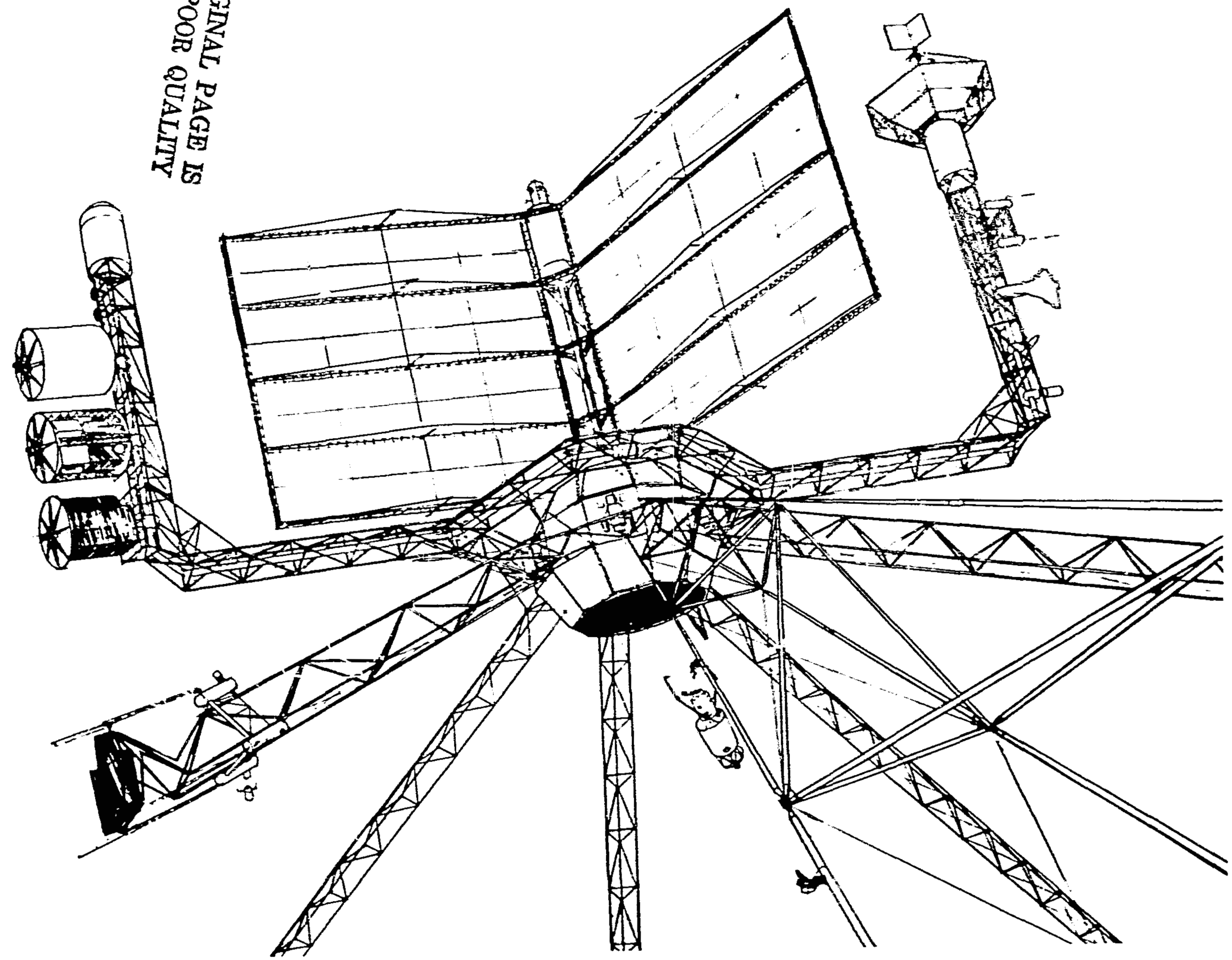

Figure 3.10.8 Power Satellite Orbital Logistics and Assembly Station

3.10.1.3.3 Operational Constraints

The van Allen radiation belts must be traversed on the way to geostationary orbit. The resultant degradation to solar cell arrays will impact electric tug operations if they utilize solar cells.

Solar-electric propulsion (SEPS) studies currently in progress have evaluated degradation of solar cells on SEPS transfers from low orbits to geosynchronous orbit. The amount of degradation varies with the type of cells, trip time, shielding, and other factors. Lithium-doped cells were found to be most radiation resistant, but even these experienced degradations on the order of 50 percent in a single round trip.

3.10.1.3.4 Other Factors

The transportation requirement imposed by this system includes strong economic considerations.

The key parameters in economics feasibility of power satellites are:

- The system development investment required
- The cost and weight of the satellites and their ground-based power receiving antennas
- The value of power derived from the system, i.e., competitive busbar cost
- The cost of ground and space operations associated with launch, assembly, operations, and maintenance of the systems
- The cost of transportation from low Earth orbit to geosynchronous orbit, and most significant, the efficiency of that transfer as reflected into total mass transportation requirements to Earth orbit
- The cost of transportation to low Earth orbit

Various economics studies have projected a competitive busbar cost at about \$0.025/kwh in 1975 dollars. This value, together with preliminary estimates of development and operations costs, and assumed use of electric propulsion for the geosynchronous transfer, results in the tradeoff of satellite cost and weight, and low orbit transportation cost, shown in figure 3.10-9. Observations that may be derived from the figure are that:

- The satellites must be producible, including costs of orbital assembly, at costs per unit weight (i.e., dollars per pound), comparable to those experienced for commercial or military jet aircraft. Since the satellites will be large and simple structures compared to aircraft, this appears feasible provided that orbital assembly costs can be kept within reasonable bounds.

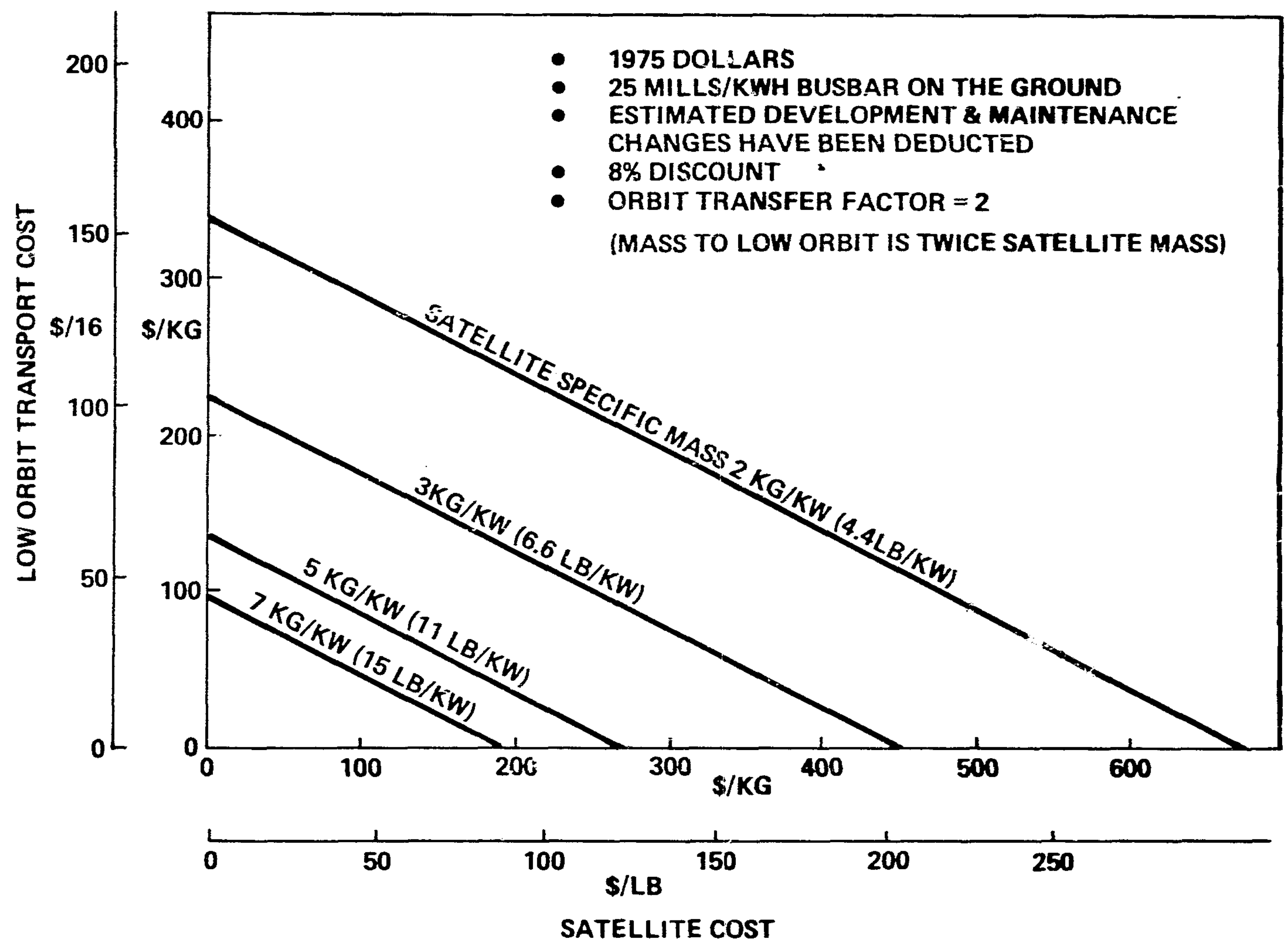

Figure 3.10-9. Power Satellite Economics Overview

- Cost of low orbit transportation must be in the range \$20 \$100/kg (\$10 \$45/lb). Payload per flight should be large to aid in minimizing orbital assembly costs.
- Economic feasibility of a total satellite energy system presents challenges to technology and systems development but on the basis of available data appears within reach.

3.10.1.4 Mission/Transportation Modes and Operations

3.10.1.4.1 Transportation Options

A low cost heavy lift vehicle is clearly needed. A variety of concepts could be considered. A representative LCHLV concept was adopted from current Boeing IR&D as an example of the class of capability required. This vehicle is a totally reusable VTOVL vehicle with 200 000 kg (440,000 lb) payload per launch. A typical configuration is shown in figure 3.10-10. The vehicle is water-recoverable, using an artificial fresh-water lake at KSC. The LCHLV is discussed in the appendix.

A wide variety of orbit transfer system options are possible, including:

- Chemical (LO_2/LH_2) expendable
- Chemical (LO_2/LH_2) reusable
- Separate electric propulsion using resistojets, magnetoplasmadynamics, or ion jets
- Parasite electric propulsion using the same propulsion technology options but drawing power from the power satellite modules on the way up (applicable to thermal engine power satellites only); self-propelled down for reuse by either electric or chemical propulsion
- Various degrees of expendability of thrusters, tanks, power conditioners, etc.

3.10.1.4.2 Representative Transportation Mode and System

3.10.1.4.2.1 Transportation Sequence and Characteristics

Since a large-scale orbital assembly process will be used, the low Earth orbit and orbit transfer segments of the system are not as rigorously linked as in other missions. LCHLV flights will arrive at the orbit assembly and logistics station in low orbit approximately daily. As major power satellite subassemblies or modules are completed, they are released from the assembly facility to begin the transfer to geosynchronous orbit. The sequence shown in figure 3.10-11 is a representative option employing parasite electric propulsion for the up trip and electric self-propelled propulsion down.

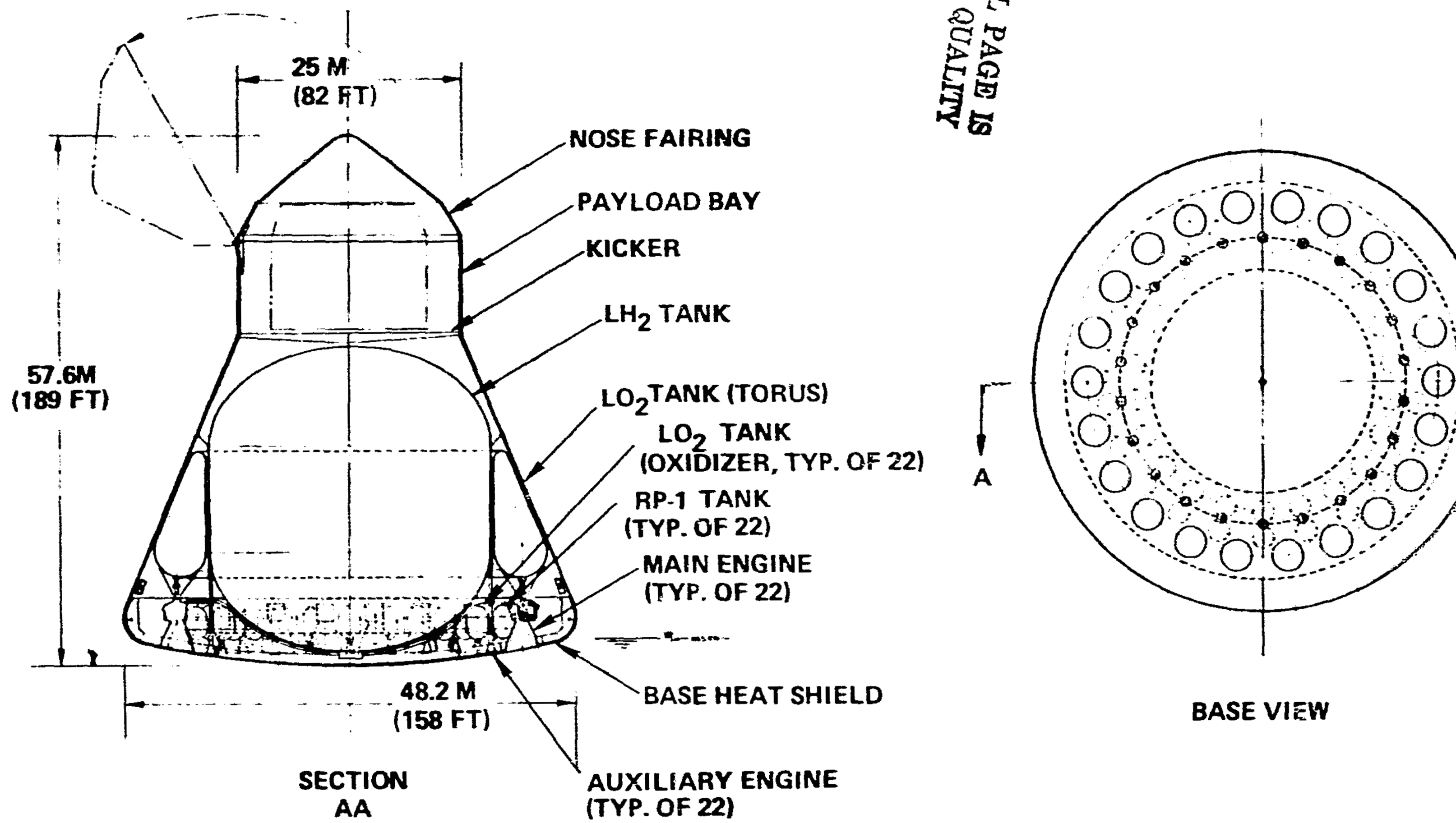

Figure 3.10-10. Low Cost Heavy Lift Vehicle Concept

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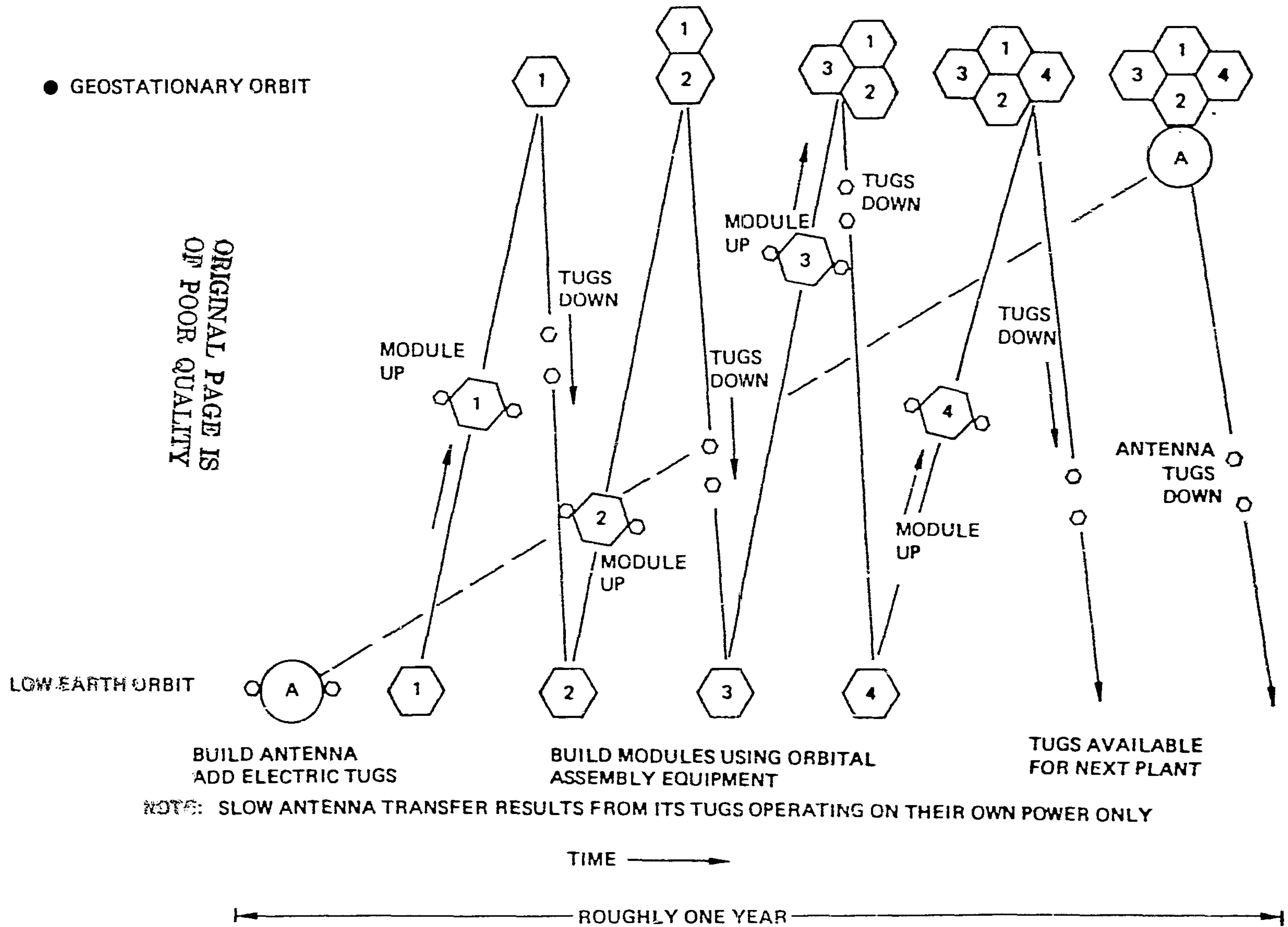

Figure 3.10.1 1. Typical Transportation Sequence for Power Satellite

The sequence illustration would not change for the separate propulsion option applicable to photovoltaic power satellites; up trip times would be greater.

A typical mass, delta V, and time history is shown in table 3.10-2. System characteristics are based on the nuclear-electric tug definition given in the appendix, with an assumed specific impulse of 36 000 m/sec (3,670 sec). The mission is delivery of a 450 000 kg (10^6 lb) powersat module to geosynchronous orbit.

Table 3.10-2. Power Sat. Mission/Transportation Sequence Mass History

Item/event	Cumulative time (days)	Delta V		Mass remaining	
		m/sec	ft/sec	10^3 kg	10^3 lb
Initial mass	0			1,286	2,835
Ascent	160	6,700	22,000	1,026	2,260
Separate payload	160			576	1,270
Jettison up propellant tanks	160			550	1,213
Descent	230	6,700	22,000	450	1,213
Down propellant tanks				(10)	995
Nuclear-electric tug				(440)	(970)

The complete sequence, for the advanced technology photovoltaic satellite of NASA CR-2357, would emcompass 25 trips such as summarized in table 3.10-2. If a fleet of 25 tugs is available, 381 days passes from initiation of ascent operations until the first satellite is completely delivered. After that, approximately 1.6 satellites per year can be delivered as shown in figure 3.10-12.

3.10.1.4.2.2 Transportation System Sizing and Performance

A highly complex tradeoff exists among several factors, including:

- The cost per unit mass of low orbit transportation.
- The unit weight (kg or lb per kw) of the power satellites.
- The unit weight and Isp of the orbit transfer system.
- Capability of the power satellite to provide power for the up transfer.
- The cost of up transfer trip time in terms of delay cost between time of investment in the satellite and beginning of its revenue return.
- The cost of investment in the orbit transfer system fleet.

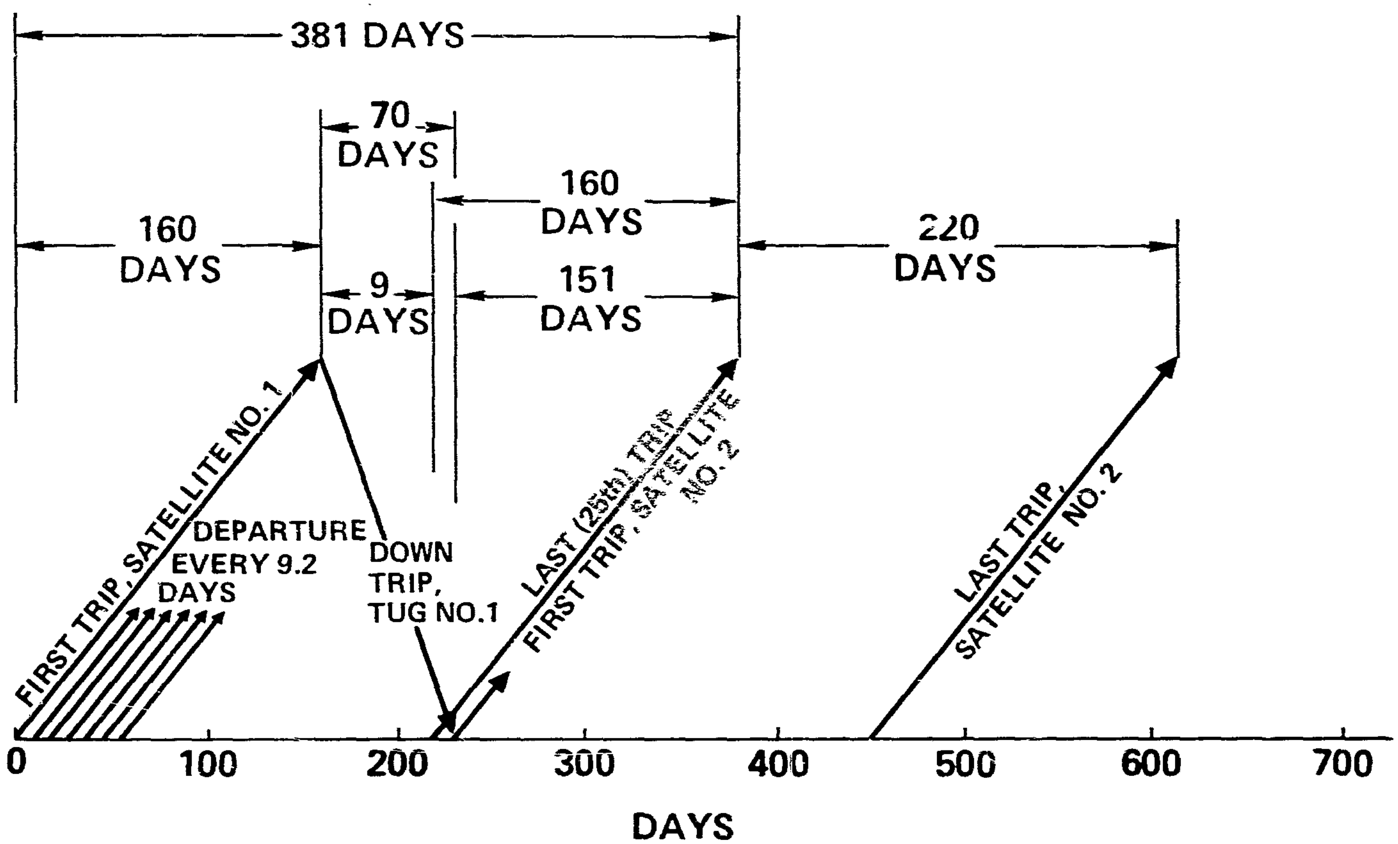

Figure 3.10-12. Power Satellite Orbit Transfer Sequencing

This tradeoff is presently under investigation. Certain principal trends are recognizable:

- The cost of transportation to low orbit is a dominant parameter. It is affected primarily by power satellite specific mass, the orbit transfer multiplier, i.e., ratio of (total mass to low orbit)/(powersat mass) and the specific cost of low orbit transportation. These parameters are nomographed in figure 3.10-13. Advanced technology power satellites and the power relay satellite may be transportable by chemical propulsion to geosynchronous orbit, while near technology satellites will probably need electric propulsion.

- In most cases it does not pay to return up propellant tanks to low orbit for reuse.

- Delay cost influences the selection of electric propulsion specific impulse. Additional factors are the cost of low orbit transportation, satellite specific mass, and electric propulsion specific mass. Over the range of expected values for these parameters, preferred specific impulse may range from 10 000 to 40 000 m/sec (1,020 to 4,100 sec). Some example results are shown in figure 3.10-14. These are selected from a wide range of options analyzed by a Boeing IR&D study. The values shown represent comparatively heavy (early technology) satellites and low costs for Earth-to-orbit transportation. The trend for these cases is to prefer an Isp of about 20 000-25 000 m/sec (2,000-2,550 sec). The trend correlation is as follows:

High Isp	Low Isp
• Heavy satellites	Light satellites
• Low cost satellites	Expensive satellites
• Higher cost transportation to low orbit	Low cost transportation to low orbit

- It is advantageous to be able to use power satellite module output to drive the up trip propulsion system, and if this is done, to use a quick (1/2 day) chemical OTV return of reusable electric thrusters and power conditioning.

The performance data described for electric systems range from an orbit transfer multiplier of 1.5 to values greater than the representative LO_2/LH_2 value of 3.5. High values result from trying to reduce trip twice below values practical for the system considered. Clearly it does not make sense to select an electric system poorer in performance than the LO_2/LH_2 alternative.

Sizing as such, i.e., selecting a particular power, size, or weight of transportation system, was not addressed. Clearly, the power satellite can be divided into subassemblies or modules of somewhat

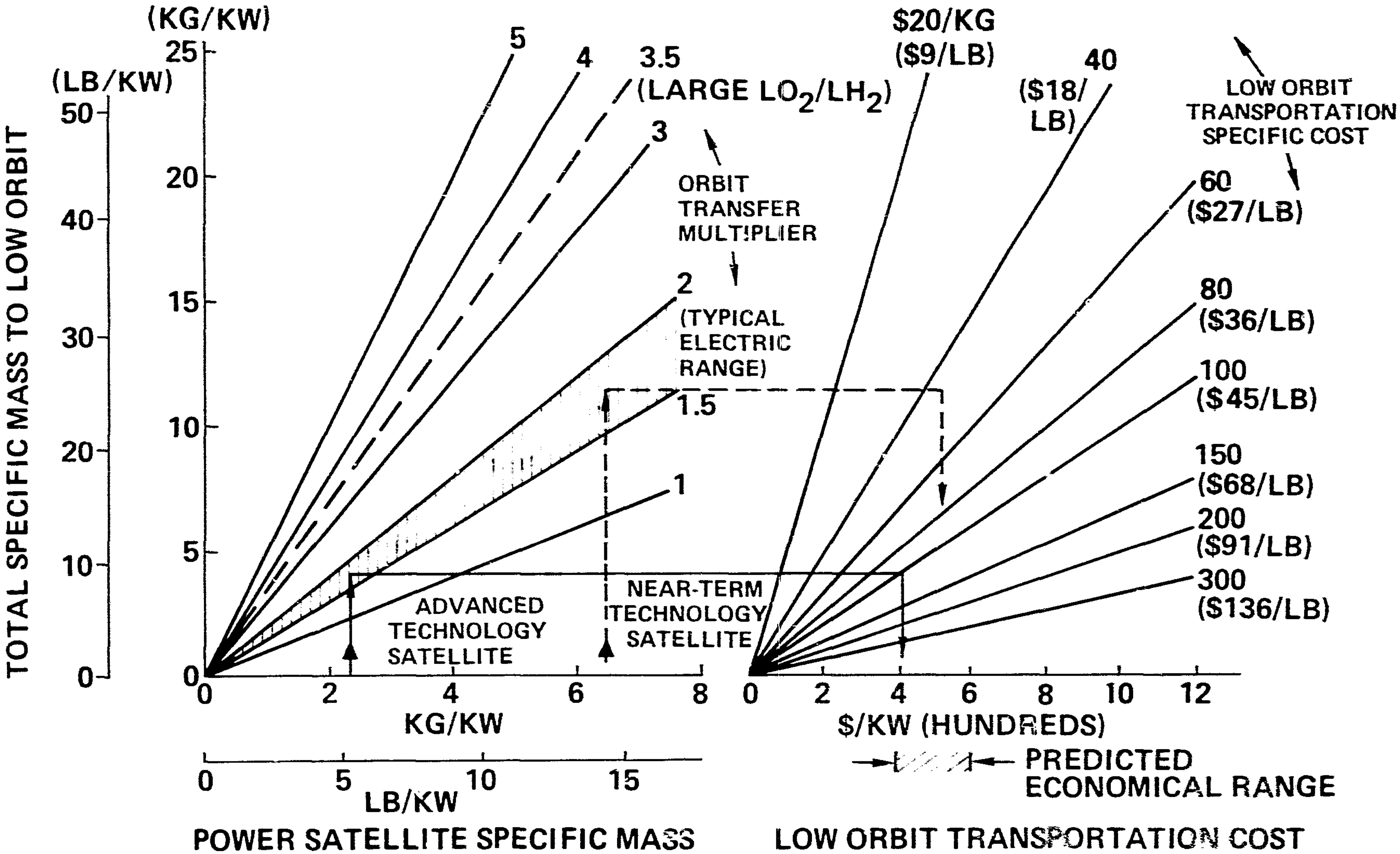

Figure 3.10.13 Low Orbit Transportation Cost Parametrics

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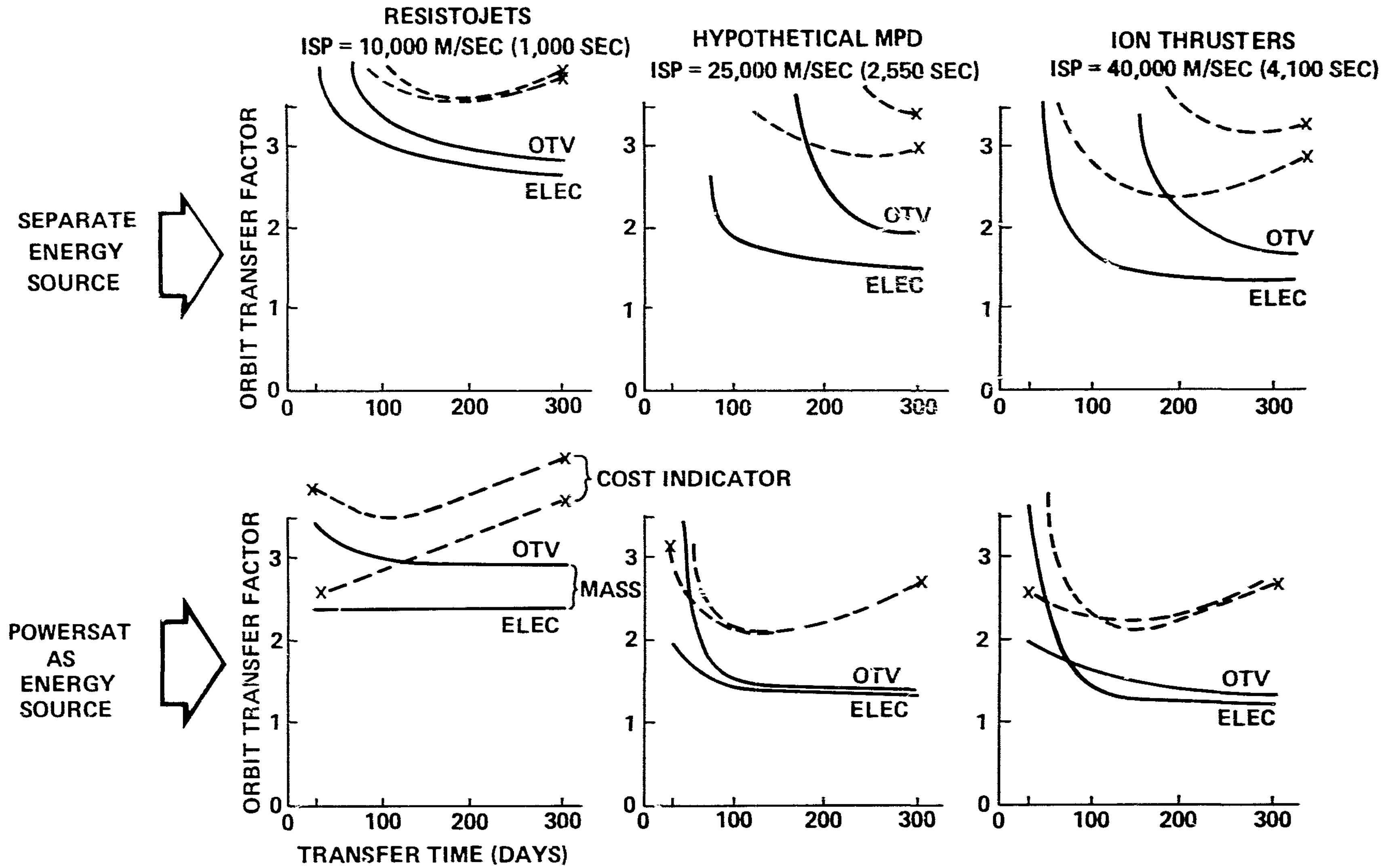

ORBIT TRANSFER FACTOR IS (TOTAL MASS IN LOW ORBIT)/(MASS OF SATELLITE)
COST INDICATOR INCLUDES FACTORS FOR BOOST COST, TRANSFER COST, AND DELAY COST

Figure 3.10-14 Comparison of Orbit Transfer Alternatives

arbitrary size. However, large modules appear to be indicated because:

- Assembly operations in geosynchronous orbit are minimized;
- Efficiency, particularly of nuclear/electric systems, is improved in large systems;
- If power from the satellite is to be employed, more or less complete modules must be handled, probably with masses in the millions of kg (lb) and electric outputs from hundreds to thousands of megawatts (the electric propulsion system need not use the total output).

3.10.1.4.2.3 Operational Factors

The nature of system operations for a power satellite transportation program is not completely understood. Studies to date have tended to deal with the problem on a piecemeal basis. Clearly, a large scale orbital assembly operation will be required.

Several to many satellite modules will be in transit to geosynchronous orbit at any one time, assuming electric propulsion. All will be under control from the ground or from the low orbit station. Capability for manned visits to these modules in transit, or in geosynchronous orbit, will be needed. A geosynchronous manned station may be required to provide for assembly and maintenance crews. Each operational satellite will require some degree of ground control and monitoring. Operational support requirements of a power satellite program are believed to imply a low orbit manned program comparable to the space base plus the geosynchronous manned station.

3.10.1.4.2.4 Earth Launch Requirements

The Earth launch system is relatively uncoupled from the orbit transfer system by low orbit assembly operations. Its size will presumably be selected based on economics considerations. These, as presently understood, seem to favor large VTOVL totally reusable "low cost heavy lift (LCHLV)" vehicles (see discussion in the appendix). Payload densities for deployable satellite elements may be as low as 20-30 kg/m^3 (1.2-2 lb/ft^3). Large payload volumes are therefore indicated. Projected launch rates are on the order of one per day for power satellites and one per week to 10 days for the power relay satellite. Figure 3.10-15 shows launch rate requirements in monograph form.

3.10.1.4.3 Transportation Options Comparison and Evaluation

An evaluation is quite difficult at the present state of knowledge. The options evaluated appear practical if they satisfy economic requirements.

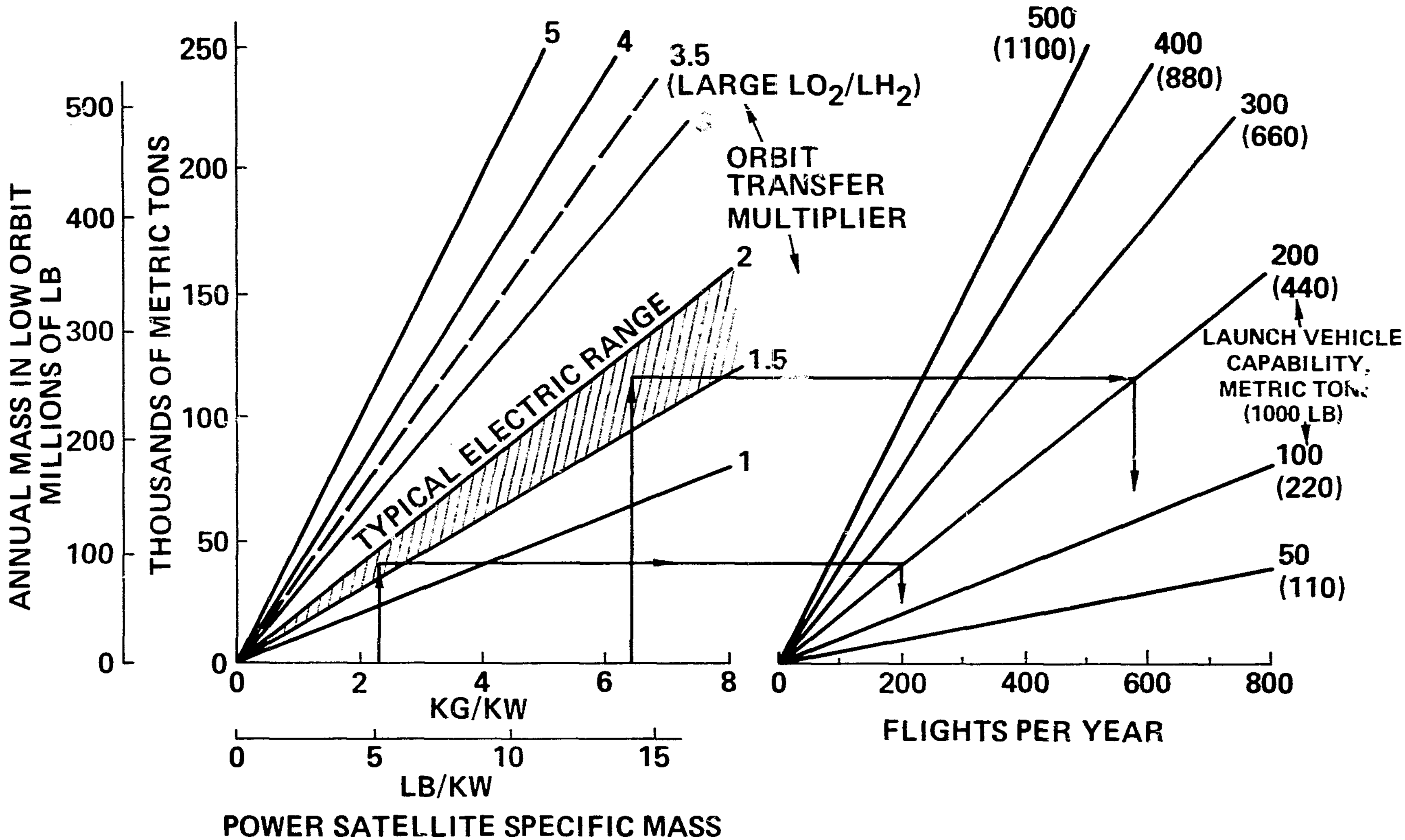

Figure 3.10-15 Low Orbit Launch Rate Requirements

3.10.2 POWER RELAY SATELLITES

Power relay satellites were not analyzed in detail. Figure 3.10-16 shows a typical relay satellite size; the mass is typically 3×10^5kg (660,000 lb). Transportation requirements are sufficiently comparable to those of the geosynchronous manned station that the transportation options and sequences described in para. 3.2.1 are applicable.

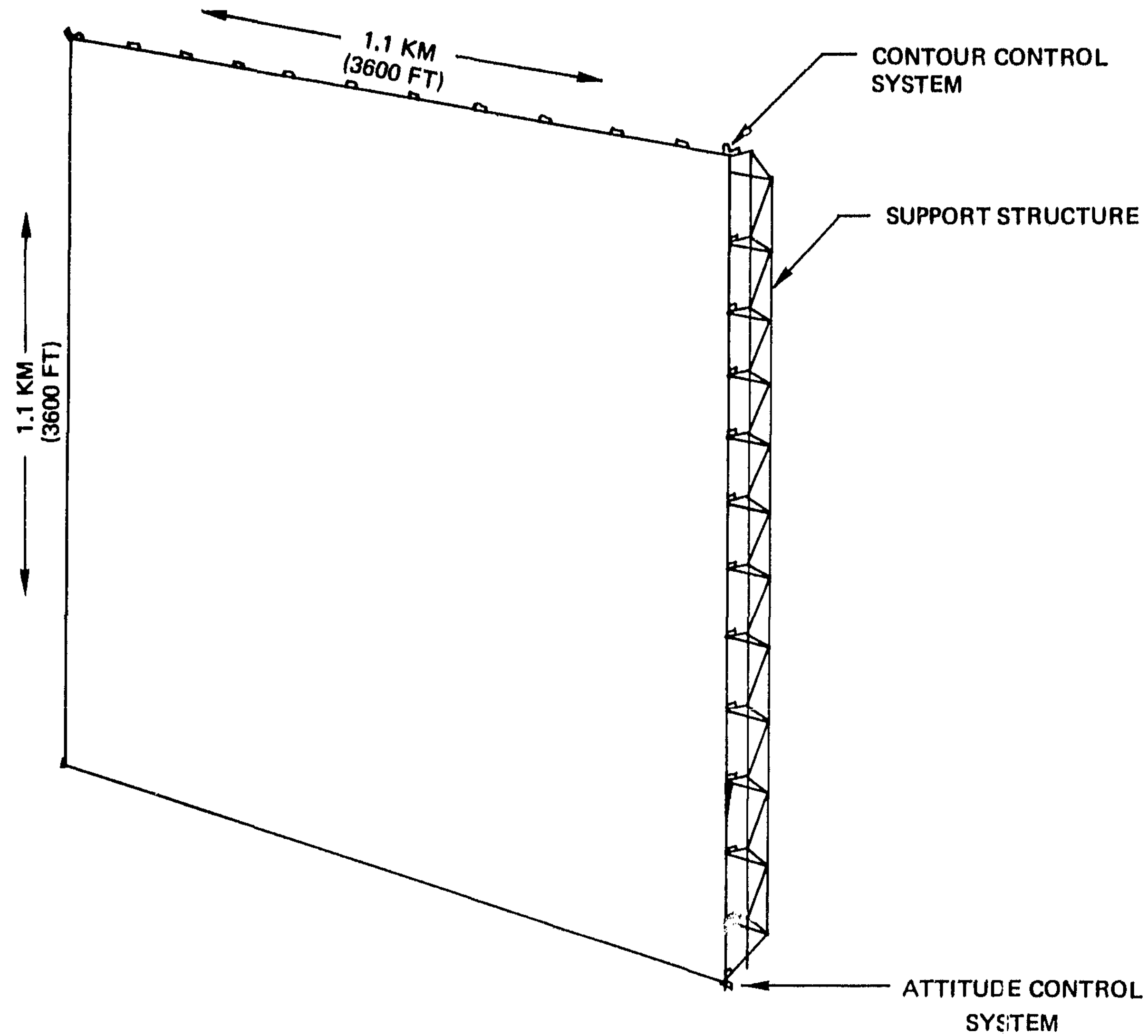

Figure 3.10-16. Passive Microwave Reflector

4.0 REFERENCES

Principal references are listed by number. Other references are also shown.

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	3.1-2	Contract NAS9-9953 MSC-00741 "Nuclear Reactor Powered Space Station" January 1971 – NAR
		Contract NAS9-9953 MSC-00721 "Space Base" July 1970 – NAR
Geosynchronous Operations		Contract NAS9-9953 MSC-00737 "Special Emphasis Tasks" July 1970 – NAR
		Contract NAS9-12909 N73-28866 "Geosync. Platform Definition Study" June 1973 – NAR
Independent Lunar (4 men)		Contract NAS8-20262 "MIMOSA" February 1967 – LMSC
Orbiting Lunar Station (8 men)	3.4-1	Contract NAS9-10924 MSC-02687 "OLS Phase A Feasibility Study" April 1971 – NAR
		Contract NAS9-9953 MSC 00737 "Lunar Orbit Space Station July 1970 – NAR
Lunar Base (12 men)	3.5-1	Contract NAS8-26145 SD71-477; "Lunar Base Synthesis Study;" May 1971 – NAR
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